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OPERATIONAL SUPPORT PLAN FOR THE REAL-TIME AUXILIARY COMPUTING FACILITY APOLLO 7 FLIGHT ANNEX

Flight Analysis Branch

MISSION PLANNING AND ANALYSIS DIVISION

MANNED SPACECRAFT CENTER

HOUSTON, TEXAS

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PROJECT APOLLO

OPERATIONAL SUPPORT PLAN FOR THE REAL-TIME AUXILIARY COMPUTING FACILITY APOLLO 7 FLIGHT ANNEX

By Ronald D. Davis
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NASA/MSC
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Mission Operations Section
TRW Systems Group

September 6, 1968

MISSION PLANNING AND ANALYSIS DIVISION
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

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DEFINITIONS

Blackout Data	Blackout data include the ground elapsed time, latitude, and longitude of each entrance and exit of the blackout region for both S-band and VHF.
Entry Data	Entry data are generally intended to imply inertial velocity, flight-path angle, latitude, and longitude at the 400,000-foot altitude, blackout data, maximum g-load, landing point, and footprint data.
Entry Interface	Entry interface is considered to be the dividing line between the tenable atmosphere and space and is considered to occur at an altitude of 400,000 feet.
Footprint Data	Footprint data include the latitude and longitude for both a zero and a full lift entry.
Horizon Monitor	Horizon monitor refers to a spacecraft attitude which is maintained 31.7 degrees below the visual horizon.
Maneuver Data	Maneuver data consist of those quantities which are necessary to completely define a maneuver. These quantities may include engine ignition, imparted velocity, guidance mode, etc.
Radar Data	Radar data include the ground elapsed time of acquisition and loss, minimum and maximum elevation angles, minimum range, tracking duration, and revolution number of each pass over a station.
RTCC State Vector	The term "RTCC State Vector" is meant to include the following quantities transmitted to the RTACF from the RTCC. 1. Vector identification 2. Lift-off time in GMT 3. Vector time in GMT 4. Position vector components 5. Velocity vector components 6. Revolution number 7. Spacecraft weight
REFSMMAT	REFSMMAT relates the Besselian coordinate system to the IMU stable member coordinate system.

NOMENCLATURE

ABDP	Apollo Block Data Program
ACRA	Atlantic continuous recovery area
ARRS	Apollo Real-Time Rendezvous Support Program
ARS	Apollo Reentry Simulation Program
CLA	contingency landing area
CM	command module
CMC	command module computer
CSM	command service module
DRA	discrete recovery area
ECS	environmental control system
EMS	entry monitoring system
EST	Eastern standard time
ETR	Eastern test range
FDO	Flight Dynamics Officer
GEMMV	General Electric Missile and Satellite Simulation Multivehicle
g. e. t.	ground elapsed time
GMT	Greenwich mean time
GNCS	guidance and navigation control system
GOST	guidance optical support table
GPMP	general purpose maneuver processor
GRR	guidance reference release
IGM	iterative guidance mode
IMU	inertial measurement unit
LES	launch escape system
LET	launch escape tower

NOMENCLATURE (Continued)

LM	lunar module
LVLH	local vertical-local horizontal coordinate system
MPT	mission plan table
MRS	Mass Properties, Reaction Control System, Service Propulsion System Program
MSFN	Manned Space Flight Network
MTVC	manual thrust vector control
NCC	corrective combination maneuver
NSR	coelliptic sequence maneuver
PAO	Public Affairs Officer
PLA	primary landing area
PUGS	propellant utilization and gauging system
PVT	pressure, volume, temperature
RCS	reaction control system
REFSMMAT	reference to stable member matrix
REM	Roentgen-equivalent-man
RTACF	Real-Time Auxiliary Computing Facility
RTCC	Real-Time Computer Complex
S-IB	first stage of Saturn IB launch vehicle
S-IVB	second stage of Saturn IB launch vehicle
SLA	spacecraft-lunar module adapter
SM	service module
SPAN	Solar Particle Alert Network
SPS	service propulsion system
SST	star sighting table
TFF	time of free fall

NOMENCLATURE (Continued)

TPF	terminal phase finalization
TPI	terminal phase initiation
VHF	very high frequency

OPERATIONAL SUPPORT PLAN FOR THE

REAL-TIME AUXILIARY COMPUTING FACILITY

APOLLO 7 FLIGHT ANNEX

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1. INTRODUCTION

1.1 Purpose

The Apollo 7 Flight Annex briefly summarizes those aspects of the operational support plan peculiar to the Apollo 7 mission. It is primarily intended for use by those individuals assigned to the Real-Time Auxiliary Computing Facility (RTACF) Apollo 7 Operational Support Team. It is intended to provide a central source of RTACF mission support information pertaining to this particular mission.

1.2 Method of Presentation

The Apollo 7 Flight Annex is divided into seven sections, the first of which is the Introduction. The second section states the main objectives of the mission and briefly outlines the major nominal mission events. The third section contains a brief description of the RTACF computing requirements for mission support, while the fourth section presents a description of the processors that will be used to satisfy these requirements. The fifth section presents the RTACF nominal mission timeline, and the sixth section lists the personnel assigned to the RTACF in support of the Apollo 7 mission. The seventh section contains a list of references which were consulted directly in preparing this document.

2. GENERAL MISSION DESCRIPTION

2.1 Introduction

A brief description of the Apollo 7 mission profile is presented in this section. The nominal mission description and certain tables of discrete events have been extracted in part from the Apollo 7 Spacecraft Operational Trajectory (Reference 1).

2.2 Nominal Mission

Apollo 7 will be the first manned Apollo mission and will demonstrate command service module (CSM) operations, and the capability of the spacecraft, crew, and Manned Space Flight Network (MSFN) support facilities to conduct an open-ended, earth orbital mission of up to 11 days duration. In addition to fundamental spacecraft checkout and crew operations, Apollo 7 will qualify the CSM forward heat shield (flat apex area), determine the environmental control system (ECS) radiator performance and amount of coating degradation, verify proper deployment of the spacecraft lunar module adapter (SLA) panels, perform transposition and simulated docking, and evaluate CSM-active rendezvous activities. A list of nominal mission events is given in Table I.

The ground stations available for support of this mission, their tracking capabilities, and site locations were obtained from Reference 2 and are summarized in Table II. Probable locations for the four Apollo tracking ships are also included.

For the nominal trajectory simulation, Apollo 7 is assumed to be launched at 1500 hours Greenwich mean time (GMT), (10 A.M. EST), September 20, 1968, from launch complex 34 on a flight azimuth of 72 degrees from true North. The mission profile has been divided into four periods of activity.

2.2.1 First period of activity. - The first period of activity extends from lift-off to 1:02:30 (day:hr:min) ground elapsed time (g.e.t.) and consists of the following events:

a. Launch and insertion into earth parking orbit: The Apollo 7 launch vehicle is the Saturn S-IB. The launch phase consists of the complete burns of both the S-IB first stage and S-IVB second stage. The spacecraft is inserted into an orbit with apogee and perigee altitudes of 153 and 123 nautical miles, respectively. The sequence of events during launch is given in Table III.

b. Earth parking orbit coast and orbital safing of the S-IVB: The activities of the spacecraft, crew, and ground during the two revolutions of the earth parking orbit phase of the mission will consist of systems checkout, inertial measurement unit (IMU) alignment, cabin purge, and S-IVB attitude control by the crew. As the S-IVB/CSM approaches the U. S. mainland at the end of the first revolution, the remaining S-IVB propellants and cold gases are dumped through the engine.

c. Transposition and simulated docking with the S-IVB: At approximately 2:55 (hr:min) g. e. t. the CSM will separate from the S-IVB over Hawaii. After translating forward approximately 50 feet from the S-IVB, the spacecraft will assume a retrograde attitude and will fire the service module reaction control system (SM RCS) thrusters to eliminate the relative velocity between the two vehicles. In this position, the crew will photograph the deployed SLA panels.

d. SM RCS phasing maneuver: At approximately 3:20 (hr:min) g. e. t. over Antigua, the CSM will make a horizontal retrograde RCS maneuver to set up the proper phasing conditions ahead of the S-IVB for rendezvous maneuvers planned to start about 23 hours later. The maneuver will account for the differential drag between the two vehicles and will place the CSM at least 15 nautical miles ahead of the S-IVB at the time of the first service propulsion system (SPS) burn on the next day.

2.2.2 Second period of activity. - The second period of activity lasts from 1:02:30 to 3:23:00 (day:hr:min) g. e. t. and consists of the following CSM-active rendezvous maneuvers:

a. First SPS corrective combination (NCC_1): The CSM will perform the first SPS maneuver at 1:02:24:55 (day:hr:min:sec) g. e. t. when the CSM will nominally be leading the S-IVB by about 73 nautical miles. This 209-foot per second burn is performed at a pitch attitude of 75 degrees below the local horizontal. It is the first maneuver of a two-impulse transfer which will result in a desired phase angle of 1. 32 degrees and an 8-nautical mile Δh offset. This maneuver will produce the concentric setup behind and below the S-IVB and will also correct any small out-of-plane angle that may have built up.

b. Possible corrective maneuver (NCC_2): Because of probable errors in the calculation and execution of NCC_1, it is almost certain that some corrective maneuver will have to be performed prior to the coelliptic maneuver. This maneuver is scheduled to be performed over Ascension Island at approximately 27:30 (hr:min) g. e. t. to take advantage of additional U. S. tracking.

c. SPS coelliptic maneuver (NSR): At 1:03:59:56 (day:hr:min:sec) g. e. t. the CSM will approach the desired concentric situation which is nominally 82.9 nautical miles behind and 8 nautical miles below the S-IVB. The coelliptic maneuver will nominally be the second SPS burn producing a ΔV of 186 feet per second, performed in a retrograde attitude, pitched up 62 degrees from the local horizontal. Onboard optical tracking will commence after this maneuver so that the CSM, independent of ground support, can compute the upcoming terminal phase burns.

d. Terminal phase initiation (TPI) and midcourse corrections: About 82 minutes after NSR (1:05:22 (day:hr:min) g.e.t.), the TPI burn will be performed when the angle between the line of sight to the S-IVB and the local horizontal reaches 27.45 degrees and the range between the two vehicles is about 15 nautical miles. The maneuver will be calculated onboard using a transfer time of 2100 seconds. Any error arising from the execution of the TPI maneuver will be calculated onboard and will be corrected at 14 minutes and 21 minutes after TPI.

e. Terminal phase finalization (TPF) and separation maneuver: When the CSM is about one nautical mile from the S-IVB, the optical braking approach will begin using the SM RCS. Station keeping at 100 to 200 feet will begin about 7 minutes later as the vehicles pass over Hawaii. After approximately 20 minutes of station keeping, the CSM will leave the vicinity of the S-IVB by applying a 2-foot per second posigrade RCS maneuver.

2.2.3 Third period of activity. - The third period of activity, from 3:23 to 10:20 (day:hr) g.e.t., consists of the following SPS maneuvers in addition to the various system checks to be performed:

a. Third SPS burn: The third SPS burn will occur during the 58th revolution over Canarvon, 3:19:42 (day:hr:min) g.e.t. The burn is performed under stabilization and control system (SCS) control for 5 seconds to impart a ΔV of 116 feet per second. The burn is designed to establish a specific SPS oxidizer level after the burn, such that a test of the propellant utilization and gauging system (PUGS) can be performed during the fifth SPS burn. This test will be conducted by enabling two PUGS sensors to be uncovered during the fifth SPS burn.

b. Fourth SPS burn: The fourth SPS burn is a minimum-impulse burn performed over the eastern test range (ETR) at 5:00:52:02 (day:hr:min:sec) g.e.t. during the 77th revolution. The 15-foot per second maneuver, including a two-jet ullage, is performed in plane and posigrade under guidance and navigation control system (GNCS) control.

c. Fifth SPS burn: The fifth SPS burn is a 1465-foot per second burn, preceded by a two-jet ullage, and is targeted to achieve a 95 by 241-nautical mile orbit. The maneuver will occur at 6:21:07:49 (day:hr:min:sec) g.e.t. over the ETR during the 104th revolution. It will be initiated under GNCS control and will change to manual thrust vector control (MTVC) after 30 seconds.

d. Sixth SPS burn: The sixth SPS burn is the second minimum impulse burn and is performed over the ETR during the 132nd revolution at 8:18:13:15 (day:hr:min:sec) g.e.t. The maneuver is preceded by a two-jet ullage and is performed in plane and in a retrograde direction under GNCS control.

e. Seventh SPS burn: The seventh SPS burn will occur at 9:23:04:37 (day:hr:min:sec) g.e.t. during the 151st revolution over the ETR. The 203-foot per second maneuver is under SCS control and is preceded by a four-jet ullage. It is targeted to attain a desired geographic location of perigee to insure a landing in the primary recovery area.

2.2.4 Fourth period of activity. - The fourth period of activity (10:20:00 until 11:00:00 (day:hr:min) g.e.t.) consists of the deorbit and subsequent landing and recovery. The eighth SPS burn, which deorbits the spacecraft, is a 10.2-second burn in the GNCS mode. The burn will occur in the 163rd revolution at 10:19:40:04 (day:hr:min:sec). Approximately 13.8 minutes will remain for the flight crew to verify deorbit conditions, separate from the SM, and orient the command module (CM) to the entry attitude. The deorbit maneuver is initiated with the spacecraft in a retrograde attitude and pitched down 31.7 degrees below the line-of-sight to the horizon. This attitude will allow the flight crew to verify the proper deorbit attitude at SPS ignition and to manually take over control of the spacecraft should the GNCS fail to function properly. Touchdown will occur at 67 degrees west longitude and 29.8 degrees north latitude in the 164th revolution.

Table I. Nominal Mission Events

Mission Event	Time from Lift-off (day:hr:min:sec)	Geodetic Latitude (deg:min:sec)	Longitude (deg:min:sec)	Altitude (n mi)
S-IVB/CSM insertion	00:00:10:04	31:30:33N	62:22:36W	128
S-IVB/CSM separation	00:02:55:00	13:20:26N	163:47:39W	130
Begin SM RCS phasing maneuver	00:03:20:00	24:44:57N	60:48:41W	132
End phasing maneuver	00:03:20:19	24:17:16N	59:33:45W	132
First SPS ignition (NCC_1)	01:02:24:55	29:19:60S	107:52:04E	161
First SPS cutoff	01:02:25:05	29:11:27S	108:32:13E	161
Second SPS ignition (NSR)	01:03:59:56	23:28:04S	104:12:26E	148
Second SPS cutoff	01:04:00:04	23:15:29S	104:43:58E	148
SM RCS ignition (TPI)	01:05:22:51	30:27:11S	55:48:29E	153
SM RCS cutoff	01:05:23:33	29:57:00S	58:49:27E	153
SM RCS ignition (TPF)	01:05:52:11	21:20:36N	159:07:22E	125
SM RCS cutoff	01:05:52:18	21:32:07N	159:33:19E	125
Third SPS ignition	03:19:42:35	19:09:05S	104:52:36E	140
Third SPS cutoff	03:19:42:40	19:17:50S	105:10:25E	140
Fourth SPS ignition	05:00:52:02	28:54:23N	79:01:39W	92
Fourth SPS cutoff	05:00:52:03	28:53:54N	78:59:35W	92
Fifth SPS ignition	06:21:07:49	31:20:20N	78:41:13W	99
Fifth SPS cutoff	06:21:08:50	31:36:18N	74:01:01W	97
Sixth SPS ignition	08:18:13:14	25:13:26N	84:20:35W	141
Sixth SPS cutoff	08:18:13:15	25:14:04N	84:18:45W	141

Table I. Nominal Mission Events (Continued)

Mission Event	Time from Lift-off (day:hr:min:sec)	Geodetic Latitude (deg:min:sec)	Longitude (deg:min:sec)	Altitude (n mi)
Seventh SPS ignition	09:23:04:37	23:52:58N	83:05:29W	90
Seventh SPS cutoff	09:23:04:44	23:41:32N	82:35:39W	90
Eighth SPS ignition (deorbit)	10:19:39:54	11:53:36N	152:59:56W	172
Eighth SPS cutoff	10:19:40:04	12:13:25N	152:26:39W	171
Entry interface	10:19:53:50	31:01:33N	98:49:53W	66
Drogue parachute deployment	10:20:04:16	27:48:22N	66:59:12W	4
Main parachute deployment	10:20:05:08	29:48:21N	66:59:12W	2
Landing	10:20:09:53	29:48:21N	66:59:12W	0

Table II. Ground Stations and Tracking Equipment for the Apollo 7 Mission

Radar Station	Call Letters	Unified S-Band				C-Band Tracking	VHF Telemetry	UHF CMD	VHF Voice	Geodetic Latitude (deg)	Longitude (deg)	Altitude (ft)
		CMD	TLM	TRK	Voice							
Grand Bahama Island	GBI					x	x	x	x	26.636350N	78.267712W	39.37
Grand Bahamas	GBM	x	x	x	x		x		x	26.632857N	80.693417W	16.40
Grand Turk Island	GTI					x				21.462889N	71.132114W	91.86
Bermuda Island	BDA	x	x	x	x	x	x	x	x	32.348103N	64.653801W	59.06
Bermuda Island	BDQ					x				32.347963N	64.653742W	62.34
Antigua	ANT					x	x	x	x	17.144030N	61.792862W	190.29
Antigua	ANG	x	x	x	x		x		x	17.016917N	61.752849W	141.08
Vanguard	VAN					x				32.700000N	48.000000W	32.81
Canary Island	CYI	x	x	x	x	x	x	x	x	27.764536N	15.634815W	567.59
Ascension Island	ASC					x			x	07.972761S	14.401695W	469.16
Ascension Island	ACN	x	x	x	x		x	x	x	07.955056S	14.327578W	1843.83
Madrid	MAD	x	x	x	x					40.455358N	04.167397W	2706.69
Madrid backup	RID	x	x	x	x					40.452982N	04.366768W	2553.15
Pretoria, South Africa	PRE					x				25.943733S	28.358489E	5334.65
Tananarive	TAN					x			x	19.000797S	47.315053E	4337.27
Carnarvon	CRO	x	x	x	x	x	x	x	x	24.907592S	113.724247E	190.29
Mercury	MER					x	x	x	x	25.000000N	125.000000E	32.81
Woomera, Australia	WOM					x				30.819728S	136.836989E	495.41
Guam	GWM	x	x	x	x		x	x	x	13.309244N	144.734413E	416.67
Honeysuckle Creek	HSK	x	x	x	x					35.584739S	148.976578E	3766.40
Canberra backup	NBE	x	x	x	x					35.402233S	148.980057E	2208.01
Hawaii	HAW	x	x	x	x	x	x	x	x	22.122092N	159.665384W	3740.16
Huntsville	HTV						x		x	25.000000N	136.000000W	32.81
California	CAL					x			x	34.582902N	120.561152W	2119.42
Redstone	RED					x				25.000000S	118.000000W	32.81
Goldstone	GDS	x	x	x	x					35.341694N	116.873289W	3166.01
Pioneer (Goldstone backup)	PIR	x	x	x	x					35.389669N	116.849062W	3376.97
Guaymas	GYM	x	x	x	x		x			27.963205N	110.720852W	62.34
White Sands	WHS					x				32.358222N	106.369564W	4041.99
Texas	TEX	x	x	x	x		x	x	x	27.653750N	97.378471W	32.81
Merrit Island	MLA					x				28.424862N	80.664406W	39.37
Merrit Island	MIL	x	x	x	x		x		x	28.508272N	80.693417W	32.81
Patrick AFB	PAT					x				28.226553N	80.599293W	49.21

Table III. Apollo 7 Launch Vehicle Operational Trajectory Sequence of Events

Ground Elapsed Time (min: sec)	Event
-0:05.0	Guidance reference release (GRR)
0:00.0	First motion
0:00.2	Lift-off signal; initiate time base 1
0:10.2	Initiate pitch and roll maneuvers
1:16.0	Maximum dynamic pressure
2:14.5	Tilt arrest
2:17.4	Level sensor activation, initiate time base 2
2:20.6	Inboard engine cutoff
2:23.6	Outboard engine cutoff; initiate time base 3
2:24.9	Separation signal
2:25.0	S-IB/S-IVB physical separation
2:26.3	J-2 engine start command
2:28.7	Ullage burn out
2:29.9	90 percent J-2 thrust level
2:36.9	Jettison ullage rocket motors
2:43.6	Jettison launch escape tower
2:48.6	Command IGM initiation
9:53.6	Guidance cutoff signal
9:53.9	Initiate time base 4 (reflects an approximate 0.2 second systems delay)
10:03.6	Orbital insertion

3. APPROVED RTACF SUPPORT REQUIREMENTS

3.1 Introduction

The requirements which the Real-Time Auxiliary Computing Facility will support for the Apollo 7 mission are given in References 3 through 7 and are briefly summarized in this section. These requirements have been assigned to the RTACF as outlined in Section 6 of the Operational Support Plan and have been discussed and mutually agreed upon by Mission Planning and Analysis Division and Flight Control Division. Details concerning inputs from the Real-Time Computer Complex (RTCC), flight controllers, and required outputs from the RTACF processors to satisfy these requirements, are given in Section 4 of this document. It should be noted that these are the requirements which were identified prior to the beginning of flight control simulations; others can and probably will be added during flight control simulations.

3.2 Launch Abort Requirements

3.2.1 Mode I abort. - In the Mode I abort region, defined from lift-off to launch escape tower (LET) jettison, determine the landing point and corresponding ground elapsed time based on a ballistic entry given an abort state vector which contains the velocity added by the launch escape system (LES).

3.2.2 Mode II abort. - In the Mode II abort region, defined from LET jettison to the time at which the landing point from a full-lift entry reaches 3200 nautical miles downrange, determine the landing point and corresponding ground elapsed time given an abort state vector and the entry lift profile.

3.2.3 Mode III abort. - The Mode III abort region is defined as the region in which a 3200-nautical mile range landing point can be realized from a retrograde SPS burn in a horizon monitor attitude. The SPS maneuver will be followed by a coast at a zero-degree bank angle to a 0.2-g deceleration and then an entry at a 55-degree bank angle. The three Mode III requirements are as follows:

a. Determine the SPS velocity increment, burn duration and ignition IMU gimbal angles to achieve a 3200-nautical mile downrange landing point given a post S-IVB/CSM separation abort vector, SPS ignition time, and the entry lift profile.

b. Determine the CM downrange landing point given a post S-IVB/CSM separation abort vector, the SPS ignition time, velocity increment, ignition IMU gimbal angles, and entry lift profile.

c. Determine the CM downrange landing point based on the post SPS burn vector and the entry lift profile.

3.2.4 Fixed ΔV abort. – The Fixed ΔV abort is essentially a Mode III abort in which a fixed velocity increment of 600 feet per second is required to achieve a landing 8800 nautical miles downrange. The Fixed ΔV abort requirements are as follows:

a. Determine the SPS ignition time, ignition IMU gimbal angles, and landing point to achieve the 8800-nautical mile downrange landing, given a post S-IVB/CSM separation abort vector, SPS velocity increment, and entry lift profile.

b. Determine the CM landing point, given a post S-IVB/CSM separation vector, SPS ignition time, ignition IMU gimbal angles, and the entry lift profile.

c. Determine the CM downrange landing point based on the post SPS burn vector and the entry lift profile.

3.2.5 Mode IV abort. – A Mode IV abort is defined as the region in which the CSM has the capability of achieving an orbit with a perigee altitude of 75 nautical miles by using an SPS burn of 2200 feet per second or less in a posigrade horizon monitor attitude. The Mode IV requirements are as follows:

a. Determine the SPS velocity increment and the ignition IMU gimbal angles required to achieve a 75-nautical mile perigee, given a post S-IVB/CSM separation vector and the SPS ignition time. The resulting orbital quantities are to be displayed in the detailed maneuver table.

b. Verify the resulting orbit after a Mode IV abort maneuver, given the SPS ignition time and the actual velocity increment imparted to the CSM.

3.3 General Orbit Phase Requirements

3.3.1 Orbital lifetime. – Determine the orbital lifetime of the vehicle (CSM or S-IVB) in days, hours, and minutes, ground elapsed time, given a state vector, vehicle weight, and vehicle aerodynamics.

3.3.2 K-factor. — Determine the atmospheric density K-factor, given the vehicle weight, drag coefficient, effective aerodynamic cross-sectional area, and two or more state vectors.

3.3.3 Flight Dynamics Officer (FDO) orbit digitals. - Compute the FDO orbit digitals, given the vehicle weight, drag coefficient, effective aerodynamic cross-sectional area, and a state vector.

3.3.4 Relative motion. – Determine the post separation, relative motion of the S-IVB or SM with respect to the CSM or CM, given the state vectors and weights of the vehicles.

3.3.5 IMU horizon alignment. - Determine the IMU inner gimbal angle required to align a horizon alignment mark on the CM window to the horizon at a selected time, given the spacecraft state vector, REFSMMAT, and the vehicle yaw and roll angles.

3.3.6 Lift-off REFSMMAT.— Determine the lift-off REFSMMAT, given the pad location, flight azimuth, time of guidance reference release, and the values of the precession and nutation angles.

3.3.7 Radiation dosage. — Given a state vector and time interval of the required computation, determine the geomagnetic parameters, the radiation dose rates (REM per hour) and cumulative radiation dose (REMS) in the CM.

3.3.8 Orbital data for the Public Affairs Officer (PAO). — Determine orbital data as required by the Public Affairs Office.

3.3.9 Ground track. — Given a state vector, determine ground track data (latitude, longitude, altitude, revolution number, azimuth, and corresponding ground elapsed time). The interval at which data are output will be specified as well as the duration of the ground track data.

3.3.10 Rendezvous radar. — Given a CSM state vector, determine the slant range, range rate, and range acceleration from the rendezvous radar located at White Sands Missile Range to the CSM. Also determine the revolution number of the pass satisfying the minimum elevation constraint, the time of overflight, and the azimuth and elevation during the overflight.

3.3.11 Solar activity. — Given solar flare data transmitted from the radio and optical telescopes in the Solar Particle Alert Network (SPAN), reduce the data to obtain graphs of the radio frequency burst profile and particle density as a function of time in the vicinity of the earth-moon system.

3.3.12 Spacecraft-to-sun alignment. — Given a REFSMMAT, determine the CSM attitude so that the liquid waste dump nozzle, the electrical power system radiator, and the environment control system radiators receive optimum heating from the sun.

3.4 Orbital Maneuver Requirements

3.4.1 Navigation vector update evaluation. — Given an RTCC tracking vector and a spacecraft telemetry vector prior to a maneuver, apply the given maneuver to the telemetry vector using command module computer (CMC) logic, and then apply the resulting accelerations to the tracking vector. Compare the postmaneuver vectors to determine if a navigation vector update is required.

3.4.2 Maneuver evaluation. — Given a state vector before and after an orbital maneuver, a REFSMMAT, the roll angle at ignition, and the ignition time, determine the actual external ΔV components and spacecraft attitude resulting from the maneuver.

3.4.3 Rendezvous and general orbital maneuvers. — Determine the series of maneuvers required to accomplish a rendezvous plan and generate the mission plan table, detailed maneuver table, and the rendezvous evaluation table. Also generate any other orbital maneuvers required to successfully complete the mission.

3.5 Command Load Requirements

3.5.1 Command module computer uplink data. — Given a set of data in engineering units to be uplinked to the CMC, determine the octal equivalent of these data in the format and with the scaling acceptable to the CMC. Conversion of the following sets of data will be required: navigation vector update, REFSMMAT, orbital external ΔV data, and deorbit external ΔV data.

3.5.2 CMC downlink data. — Given a CMC state vector in either an octal or alphanumeric format, convert the vector to engineering units in the coordinate system acceptable to the RTCC.

3.5.3 Engineering units to octal conversion. — Convert a number in engineering units to octal, given the scale factor, precision, and multiplier to be used.

3.5.4 Octal to engineering units conversion. — Given an octal number with its associated scale factor and precision, determine the equivalent number in engineering units.

3.5.5 Navigation vector update. — Given a state vector and a time to perform a navigation vector update, determine and output, in engineering units and the correct octal format, a navigation vector update for either the CMC or the S-IVB onboard computer.

3.6 Optical Sighting Requirements

3.6.1 IMU alignment. — Determine REFSMMAT given two stars, their location in the telescope or sextant field of view, and the corresponding spacecraft IMU gimbal angles.

3.6.2 Star finding. — Given the current IMU alignment and the current spacecraft IMU gimbal angles, find two stars in the field of view of the telescope and boresight, so that one star lies on the R-line and the other star lies as close as possible to the M-line of telescope reticle pattern.

3.6.3 Star location. — Determine the sextant shaft and trunnion angles required to center two stars in the sextant field of view, given the star identification, the current REFSMMAT, and the spacecraft IMU gimbal angles at a specified time.

3.6.4 Ground target sighting. — Determine the spacecraft IMU gimbal angles, time of arrival at the desired line of sight to the target, and the time and central angle of closest approach given the target location, REFSMMAT, and desired sextant configuration.

3.6.5 Celestial target sighting. — Given the celestial target location, REFSMMAT, and fixed sextant configuration, determine the spacecraft IMU gimbal angles required to center the target in the sextant field of view. Also compute the central angle and time of closest approach, the time of arrival at the line of sight to the target, and the earliest point at which the line of sight does not pass through the earth's atmosphere.

If the sextant is to be in a movable configuration, the sextant shaft and trunnion angles are to be determined, in addition to the other quantities, given the spacecraft IMU gimbal angles at the time of the sighting.

3.7 Flight Planning and Experiments Work Schedule Requirements

3.7.1 Radar data. — Given a state vector, minimum elevation angle, and any maneuvers to be performed during a specified time interval, determine the following quantities for specific radar sites: spacecraft acquisition and loss times, slant range and azimuth at acquisition, minimum slant range, maximum elevation angle, and the duration of the pass.

3.7.2 Spacecraft daylight-darkness. — Given a state vector, time interval to be considered, and any maneuvers to be performed during the interval, determine the time and spacecraft position of sunrise and sunset and terminator crossings.

3.7.3 Spacecraft moon sighting. — Given a state vector, time interval to be considered, and any maneuvers to be performed during the interval, determine the time and spacecraft position of moonrise and moonset.

3.7.4 Computed events. — Determine the orbital events (apogee, perigee, ascending node, and revolution number) and related times of these events given an initial state vector, time interval to be considered, and any maneuvers performed during the interval.

3.7.5 Landmark sighting. — Given the landmark number, time interval to perform the landmark search, and a state vector, determine the following quantities: the spacecraft acquisition and loss times, slant range and azimuth at acquisition, minimum slant range, maximum elevation angle, and the duration of the pass.

3.7.6 Spacecraft star sighting. — Given a star identification number, revolution number, and a state vector, determine the times of star rise and star set relative to the spacecraft and the time and position in which the star-earth-spacecraft central angle is a minimum (closest approach).

3.7.7 Closest approach. - Determine the spacecraft closest approach to a specified ground target, given the target identification, revolution number, and a state vector.

3.7.8 Pointing data. — Given a state vector, target identification, REFSMMAT, and time interval to perform the target search, determine the spacecraft to target look angles (gimbal angles and local vertical/local horizontal angles) and the target to spacecraft look angles (elevation angle and azimuth angle). Also compute the spacecraft acquisition and loss times, maximum elevation angle, minimum slant range, altitude at minimum slant range, and elapsed time of the pass.

3.8 CSM Systems Requirements

3.8.1 Mass properties and aerodynamics. - Given the weights, centers of gravity, and moments of inertia of the consumables tanks, determine the following quantities:

a. Aerodynamics for CM entry

b. Center of gravity locations for different CM and SM consumable and equipment configurations

c. Mass properties table for a specific SPS oxidizer to fuel mixture ratio

d. Digital autopilot command load for a specific SPS thrust level.

3.8.2 SM RCS propellant profile. - Determine the complete SM RCS propellant budget, given the spacecraft mass properties, the control mode for each maneuver, the RCS jet selection, and a timeline of maneuvers.

3.8.3 SM RCS propellant status. — Determine the current SM RCS propellant available using the primary or auxiliary system given the quad selection, the corresponding helium pressures and temperatures, the tank expulsion efficiencies, and the RCS oxidizer to fuel mixture ratios.

3.9 Deorbit Requirements

3.9.1 Primary landing area (PLA). — Determine the deorbit ignition time, IMU gimbal angles at ignition, and the time to reverse the bank angle from the initial bank angle in order to achieve a target latitude and longitude in one of the primary landing areas. The maneuver will be based on a specified attitude, REFSMMAT,* and an incremental velocity or a velocity and flight-path angle constraint at entry interface. The entry profile will consist of a lift vector orientation from 400,000 feet to a specified g-level, followed by a bank-reverse-bank angle entry to drogue chute deployment.

3.9.2 Contingency landing area (CLA). - Determine the deorbit ignition time and IMU gimbal angles at ignition to achieve a target longitude in one of the contingency landing areas based on a specified attitude, REFSMMAT,* an incremental velocity or a velocity and flight-path angle constraint at entry interface, and an entry profile. This profile will consist of (1) a lift vector orientation from 400,000 feet to a specified g-level, followed by constant bank angle to droque chute deployment, or (2) a rolling entry from 400,000 feet to drogue chute deployment.

3.9.3 CM RCS deorbit. - Determine the CM RCS deorbit ignition time and IMU gimbal angles at ignition to achieve a target longitude based on an incremental velocity, spacecraft attitude, REFSMMAT,* and a rolling entry from 400,000 feet to drogue chute deployment.

3.9.4 Hybrid deorbit. - Determine the SM and CM RCS deorbit ignition times in order to achieve a target longitude given the incremental velocities and spacecraft attitudes for each maneuver, REFSMMAT,* and an entry profile consisting of a lift vector orientation to a specified g-level, followed by a zero-lift entry to droque chute deployment. Also determine the landing point based on the actual hybrid deorbit performed. The SM RCS maneuver will be considered as having been performed nominally, and the actual incremental velocities will be used to define the CM RCS maneuver. The entry profile will be the same as that mentioned previously.

3.9.5 Hybrid deorbit without SLA separation. - Given a state vector before S-IVB/CSM separation, the incremental velocities to be added by the SM and CM RCS thrusters, the spacecraft attitude to be maintained during each maneuver, REFSMMAT,* and the interval of time required to separate the CM from the S-IVB/SM, determine the RCS ignition times and IMU gimbal angles at ignition to achieve a target longitude. The entry profile will nominally consist of a lift vector down to a 1-g deceleration followed by a rolling entry to drogue chute deployment.

3.9.6 Deorbit with separation maneuver. - Given a state vector prior to separation, a specified separation and deorbit maneuver, and an entry profile, determine the time to perform the SPS deorbit maneuver and all the entry quantities associated with the deorbit. The separation maneuver will either occur at a fixed ground elapsed time or at a fixed time interval prior to SPS ignition. The separation maneuver will be defined by the velocity increment of the burn, the spacecraft attitude, and the ignition time, which will either be given (fixed time) or will be computed (fixed Δt). The deorbit maneuver will be specified in the same manner as a PLA or CLA.

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*The REFSMMAT will either be given or computed from the spacecraft body attitude and IMU gimbal angles.

3.9.7 Block data. - Determine PLA and CLA deorbit ignition times for a series of deorbits, given a velocity and flight-path angle constraint at entry interface, the spacecraft attitude at ignition, the entry profile to be flown, and the type of landing area to be considered. The deorbit data are to be computed in blocks of six revolutions and are to reflect any planned maneuvers occurring in this period. The first set of block data should reflect the S-IVB/CSM separation maneuver. For all deorbits occurring prior to the nominal S-IVB/CSM separation time, the deorbits will be computed with the separation maneuver occurring at a fixed interval of time before deorbit ignition. All deorbits in the first block of data occurring after nominal separation time will simulate the separation maneuver at the fixed nominal time.

3.9.8 Guided entry and backup guidance quantities. - Determine the entry monitoring system (EMS) initialization quantities and the guided entry profile and backup guidance quantities required to reach a target landing point given a state vector at entry interface. The state vector at entry interface will be generated by a deorbit processor which will either simulate the deorbit maneuver given a preburn state vector or will coast a deorbit postburn vector to entry interface.

4. RTACF PROCESSORS FOR THE APOLLO 7 MISSION

4.1 Introduction

This section of the Flight Annex presents a description of the processors that will be used in the RTACF to satisfy the Apollo 7 mission requirements which have been received to date. It also presents examples of the types of computer output available in the RTACF.

4.2 Types of RTACF Output Available

There are two types of output available for the Apollo 7 processors to be used in the RTACF. The first type of output is in the form of a summary sheet which is designated as the on-line computer output, while the second type is printed off-line from a computer magnetic tape written by the processor. The summary sheets, which are output immediately after the termination of the computation, display at least the minimum number of output parameters necessary to satisfy the requirements. The off-line computer printout can be obtained if desired and contains additional information to supplement the on-line summary sheets. Table IV is a list of all the available data which can be obtained from the off-line computer printout.

There are presently 31 different on-line summary sheets available in the RTACF. The standard summary sheet will be used to output the results of 13 of the RTACF processors, and the FDO detailed maneuver table will be used for an additional 4 of the processors. The standard summary sheet has been formatted to include selected outputs from the 13 processors. Even though the outputs of these processors are somewhat similar, no single processor is capable of providing all the data indicated on the standard summary sheet. Thus, in order to interpret the standard summary sheet correctly, some detail of the outputs of each of the particular processors must be known. However, for any processor using the standard summary sheet, all of the required data for that processor will be included.

The remaining 29 summary sheets are formulated specifically for particular processors or functions. The details of the processors associated with each summary sheet are described in the remainder of this section.

4.3 Launch Abort Processors

Five abort regions have been defined for the Apollo 7 launch phase. The fulfillment of the requirements for these regions necessitated the development of nine RTACF abort processors. Two processors were developed for the Mode I region, one for each of the Mode II and Fixed ΔV regions, three for the Mode III region, and two for the Mode IV region. The five launch-abort modes and the processors required to support each mode are described below.

4.3.1 Mode I abort. - The Mode I abort region is defined from lift-off until LET jettison which occurs 15 seconds after S-IVB ignition. The Mode I landing area is completely contained in the Atlantic Continuous Recovery Area (ACRA) which extends downrange 3200 nautical miles along the flight azimuth.

Mode I abort processor 1: The actual wind profiles encountered from 24 hours to 1 hour prior to launch are employed in this processor to predict the spacecraft landing points should a Mode I abort be performed during the first 90 seconds of the flight. Selected abort times are considered in this time interval to determine which aborts will result in a land impact about the launch area.

Inputs required:

a. Wind profile from the Kennedy Space Center

b. CM entry weight

Outputs required:

The outputs required for this processor will be displayed on the Mode I launch-abort summary sheet shown in Figure 1.

Mode I abort processor 2: This processor computes the actual landing point of the CM in the Mode I abort region using the beginning of mission aerodynamics and a state vector which contains the velocity increment from the LES. The effects of winds on the landing point are not considered in this processor.

Inputs required:

a. RTCC state vector

b. CM entry weight

Outputs required:

a. Ballistic landing point (latitude and longitude)

b. Time of landing (ground elapsed time)

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 2.

4.3.2 Mode II abort. - The Mode II abort region begins at LET jettison (S-IVB ignition plus 15 seconds) and terminates when a full lift abort trajectory results in a landing range equal to 3200 nautical miles. A Mode II abort is essentially a full lift CM entry after separation from the S-IVB and involves no retrograde burn.

Mode II abort processor: The Mode II abort processor is used to determine Mode II abort landing points. In addition, it is also used for

Mode III and Fixed ΔV aborts when the input vector is an RTCC postburn vector and a constant bank-angle entry is flown from 400, 000 feet to drogue chute deployment.

Inputs required:

a. RTCC state vector which reflects the CSM/S-IVB separation maneuver (Mode II) or the SPS retrograde maneuver (Mode III and Fixed ΔV)

b. Lift profile

Outputs required:

a. Latitude, longitude, and ground elapsed time of landing

b. Maximum g-load during entry

c. Velocity, flight-path angle, and ground elapsed time at 400, 000 feet

d. Blackout data

Optional output:

Footprint data

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 2.

4.3.3 Mode III abort. - The Mode III abort region begins when a Mode II full lift abort trajectory results in a landing range greater than 3200 nautical miles. It normally terminates when the CSM enters the Fixed ΔV abort region. The normal Mode III entry profile is a lift vector up (zero-degree bank angle) to a g-load of 0.2 followed by a constant bank angle of 55 degrees south to drogue chute deployment. If in the event of a Mode III abort, this type entry results in a landing point within the ACRA, no retrograde SPS burn will be performed. However, when the trajectory results in a landing range greater than 3200 nautical miles, a retrograde SPS burn will be performed to place the landing point at the discrete recovery area (DRA) which is 3200 nautical miles downrange.

The Mode II abort processor previously described will be used to compute the landing points for those Mode III aborts which do not require a retrograde burn to place them in the ACRA and which maintain a constant bank angle from 400, 000 feet to drogue chute deployment. It will also be used to compute the landing points when an RTCC postburn vector is input and a constant bank-angle entry is flown from 400, 000 feet to drogue chute deployment.

Three Mode III abort processors will be employed to compute the landing points when retrograde burns are required and the entry profile consists of a lift vector orientation to a certain g-load followed by a constant bank angle to drogue chute deployment.

Mode III abort processor 1: This processor uses an iterative technique to determine the SPS incremental velocity needed to place the CM landing point at the DRA when the ignition time is specified.

Inputs required:

a. RTCC preburn state vector which reflects the CSM/S-IVB separation sequence

b. SPS ignition time

c. Retrograde attitude

d. Entry lift profile

e. CSM weight at CSM/S-IVB separation

f. CM entry weight

Outputs required:

a. Velocity increment needed to place the landing point at the DRA

b. Burn duration

c. IMU gimbal angles at ignition

d. IMU gimbal angles and ground elapsed time at 400,000 feet

e. Velocity, flight-path angle, latitude, and longitude at 400,000 feet

f. Blackout data

g. Maximum g-load during entry

h. Latitude, longitude, and ground elapsed time of landing

Optional outputs:

Footprint data

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 2.

Mode III abort processor 2: Processor 2 computes the landing point of the CM once the SPS ignition time, incremental velocity, and entry profile have been specified.

Inputs required:

a. RTCC preburn state vector which reflects the CSM/S-IVB separation sequence

b. SPS ignition time

c. Velocity increment

d. Retrograde attitude

e. CSM weight at CSM/S-IVB separation

f. CM entry weight

g. Entry lift profile

Outputs required:

a. IMU gimbal angles and ground elapsed time at 400, 000 feet

b. Velocity, flight-path angle, latitude, and longitude at 400, 000 feet

c. Blackout data

d. Maximum g-load during entry

e. Latitude, longitude, and ground elapsed time of landing

Optional outputs:

Footprint data

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 2.

Mode III abort processor 3: This processor is employed to determine the CM landing point when a post SPS burn state vector is supplied from the RTCC. It is also used to determine the CM landing point for the Fixed ΔV abort given the postburn vector and a similar entry profile.

Inputs required:

a. RTCC postburn state vector

b. Entry lift profile

c. CM entry weight

Outputs required:

a. IMU gimbal angles and ground elapsed time at 400,000 feet

b. Velocity, flight-path angle, latitude, and longitude at 400,000 feet

c. Blackout data

d. Maximum g-load during entry

e. Latitude, longitude, and ground elapsed time of landing

Optional outputs:

Footprint data

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 2.

4.3.4 Fixed ΔV abort. - The Fixed ΔV abort is defined as a region in which a range of 8800 nautical miles can be realized with a retrograde SPS burn of 600 feet per second and an entry consisting of full lift to 0.2 g-load and a 55-degree bank angle south to drogue chute deployment. The region is further defined by the following constraints: (1) the minimum time of free fall (TFF) to 300,000 feet must be 100 seconds; (2) the SPS ignition time can occur no sooner than 125 seconds after CSM/S-IVB separation; and (3) the flight-path angle at the time of abort must be steeper than approximately -1.5 degrees to avoid skipout conditions.

Three Fixed ΔV abort requirements exist. Two require the input of a preburn vector, and one requires a vector after the SPS burn. The preburn requirement to determine the CM landing point, when given the ignition time and incremental velocity, is identical to the requirement satisfied by the Mode III abort processor 2. Also, due to the similarity of the requirements, the Mode III abort processor 3 has been used to satisfy the Fixed ΔV postburn requirement. The remaining Fixed ΔV preburn requirement is satisfied by the following processor.

Fixed ΔV abort processor: This processor computes the time of ignition to achieve an 8800-nautical mile range landing point with a 600-foot per second SPS burn.

Inputs required:

a. RTCC preburn state vector which reflects the CSM/S-IVB separation sequence

b. Incremental SPS velocity

c. CSM weight at CSM/S-IVB separation

d. CM entry weight

e. CSM attitude at ignition

f. Entry lift profile

Outputs required:

a. SPS ignition time

b. IMU gimbal angles and ground elapsed time at 400,000 feet

c. Velocity, flight-path angle, latitude, and longitude at 400,000 feet

d. Blackout data

e. Latitude, longitude, and ground elapsed time of landing

Optional outputs:

Footprint data

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 2.

4.3.5 Mode IV abort and apogee kick. - The Mode IV abort region begins when the CSM has the capability of achieving an orbit with a perigee altitude of 75 nautical miles with a 2200-foot per second SPS burn and ends when the S-IVB has placed the CSM in an acceptable orbit. Two sequences can be employed to reach insertion. One requires a posigrade SPS maneuver immediately following CSM/S-IVB separation (Mode IV); the other requires a CSM/S-IVB separation followed by a coast to apogee before the SPS burn is performed (apogee kick).

Two Mode IV processors have been developed. One determines the SPS incremental velocity required to achieve a 75-nautical mile perigee altitude while the other is used to verify the RTCC value of ignition time and burn duration for the Mode IV maneuver.

Mode IV abort processor 1: This processor determines the incremental velocity to be added by the SPS engine to achieve an orbit with a 75-nautical mile perigee altitude. The ignition time may be either at apogee or immediately following CSM/S-IVB separation.

Inputs required:

a. RTCC preburn state vector which reflects the CSM/S-IVB separation sequence

b. Time of SPS ignition

c. CSM attitude at ignition

d. CSM weight after separation

Outputs required:

The outputs required for this processor will be displayed on the FDO detailed maneuver table shown in Figure 3.

Mode IV abort processor 2: This processor determines the orbit resulting from the actual value of incremental SPS velocity used to attain insertion.

Inputs required:

a. RTCC preburn state vector which reflects the CSM/S-IVB separation sequence

b. Time of SPS ignition

c. CSM attitude at ignition

d. CSM weight after separation

e. Velocity increment imparted to CSM

Outputs required:

The outputs required for this processor will be displayed on the FDO detailed maneuver table shown in Figure 3.

4.4 General Orbit Phase Processors

There are presently 11 processors which can be included under the classification of general orbit phase processors. From these processors a variety of information is obtained concerning the mission during the coast periods. Such data as orbital lifetime, relative motion, groundtrack, etc., are obtained exclusive of any orbital maneuvers, optical sightings, and command load computations.

4.4.1 Orbital lifetime. - This processor computes the predicted orbital lifetime of the spacecraft, given a state vector, the aerodynamic characteristics of the vehicle, and the model atmosphere to be used.

Inputs required:

a. State vector

b. Year, month, and day of launch

c. Vehicle drag coefficient and reference area

d. Atmosphere model to be used

e. Vehicle weight

Outputs required:

Lifetime in days, hours, and minutes from lift-off and from the time of the vector.

4.4.2 K-factor. - This processor computes the atmospheric density K-factor to be used in the RTCC. The K-factor value is determined by propagating one input state vector to the time of a second input state vector. The value of the atmospheric density multiplier is adjusted until the propagated vector and succeeding state vectors agree to some specified accuracy.

Inputs required:

a. Two or more state vectors for the same vehicle

b. Spacecraft weight

c. Spacecraft drag coefficient and reference area

Outputs required:

a. Value of K-factor

b. Probable error in K-factor

4.4.3 <u>FDO orbit digitals</u>. - This processor takes a state vector and computes the orbital quantities included in the RTCC FDO orbit digitals display and presents them in a similar format.

Inputs required:

a. RTCC state vector

b. Spacecraft weight

c. Threshold time or revolution number

Outputs required:

The outputs required for this processor are displayed on the FDO orbit digitals summary sheet shown in Figure 4.

4.4.4 <u>Relative motion</u>. - The relative motion processor computes the relative motion quantities associated with an active and passive vehicle. The quantities are measured relative to a curvilinear coordinate system whose origin can be centered at either vehicle.

Inputs required:

a. RTCC state vectors for the two vehicles

b. Current weight of each vehicle

c. Current drag coefficient and reference area for each vehicle

d. Vehicle at which coordinate system will be centered

e. Any maneuvers performed by the active vehicle

Outputs required:

The outputs required for this processor are displayed on the relative motion summary sheet shown in Figure 5.

4.4.5 <u>IMU horizon alignment</u>. - This processor will calculate the IMU inner gimbal angle required to align a horizon alignment mark on the CM window to the horizon at a specified time.

Inputs required:

a. RTCC state vector

b. REFSMMAT

c. Horizon monitor attitude

d. Time of computation

e. Spacecraft weight

f. IMU roll and yaw gimbal angles

Outputs required:

IMU inner gimbal angle at the specified time.

4.4.6 <u>Lift-off REFSMMAT</u>. - This processor computes the REFSMMAT that will be used from lift-off until the IMU is realigned in orbit.

Inputs required:

a. Latitude and longitude of launch pad

b. Altitude of launch pad

c. Precession and nutation angles

d. Flight azimuth

e. Time of guidance reference release

f. Month and day of launch

g. Greenwich hour angle

Outputs required:

Lift-off REFSMMAT

4.4.7 <u>Radiation evaluation</u>. - This processor takes a state vector and, at given intervals along a trajectory, determines geomagnetic parameters and the radiation dose rates in the command module. It also calculates total REM dose in the command module by integrating the radiation dose rates over a particular portion of the trajectory.

Inputs required:

a. RTCC state vector

b. Spacecraft weight

c. Spacecraft drag coefficient and reference area

Outputs required:

The outputs required for this processor will be displayed on the radiation evaluation summary sheet shown in Figure 6.

4.4.8 Orbital data for the Public Affairs Officer. - The PAO requirements are not satisfied by any single processor. The outputs satisfying the desired request will be displayed on the summary sheet of the processor used.

4.4.9 Ground track. - The ground track requirements are not satisfied by any single processor. Ground track data capability is available with any General Electric Missile and Satellite Simulation Multivehicle (GEMMV) processor by selection of the ground track option. The output satisfying this requirement will be displayed on the ground track summary sheet shown in Figure 7.

4.4.10 Solar Particle Alert Network. - The SPAN processor will process solar flare data received from the optical and radio telescopes in the Solar Particle Alert Network. The input data will be received in the form of a punched paper tape which is generated at the Space Environment Console. The SPAN processor will first transfer these data from the paper tape to a magnetic tape. It will then process the magnetic tape and display the resulting data in a form from which the radiation hazard in the earth's vicinity can be determined.

The processor is being developed for use in the Apollo high ellipse missions where the nominal orbit will take the astronauts through the earth's radiation belts. The Apollo 7 mission will be used to check out the SPAN processor and the procedures employed in transmitting the observation data to the RTACF.

At the time of this writing, the SPAN processor is in the developmental stage, and the inputs and outputs required by the processor have not been fully defined.

4.4.11 Spacecraft-to-sun alignment. - This processor computes the CSM attitude in gimbal angles necessary to expose certain areas of the vehicle to direct sunlight for heating purposes. These areas include the liquid waste dump nozzle and the electrical power and environmental control system radiators.

Inputs required:

a. Area of the spacecraft to be exposed to the sun

b. REFSMMAT

c. Right ascension and declination of the sun

Outputs required:

IMU gimbal angles required to orient the specified areas to the sun.

4.5 Orbit Maneuver Processors

The orbit maneuver requirements are satisfied by three GEMMV processors and a general rendezvous support program. Two of the processors are used to evaluate orbital maneuvers: one determines the effect of any errors in the CMC state vector on an upcoming maneuver while the other evaluates the results of a completed maneuver. The third processor can simulate a variety of maneuvers once the maneuver data has been specified. The rendezvous program is used to plan the rendezvous maneuvers and can be used to compute any other maneuvers performed during the Apollo 7 mission.

4.5.1 Navigation vector update evaluation. - This processor determines whether a navigation vector update is necessary prior to a planned maneuver. A spacecraft telemetry vector and an RTCC state vector are input and propagated to the maneuver time. The maneuver is applied to the spacecraft telemetry vector and the acceleration profile resulting from the use of the CMC guidance logic is applied to the RTCC tracking vector. After the completion of the maneuver, the vectors are compared to see if a navigation vector update is required.

Inputs required:

a. RTCC state vector

b. Spacecraft telemetry vector

c. REFSMMAT

d. Maneuver targets

e. Guidance mode

f. Spacecraft weight

Outputs required:

The outputs required for this processor are displayed on the FDO detailed maneuver table shown in Figure 3.

4.5.2 General orbit maneuver. - This processor can simulate any orbital maneuver given the appropriate burn quantities, spacecraft attitude during the maneuver, inertial platform alignment, and the type of guidance mode to be used during the maneuver.

Inputs required:

a. RTCC state vector

b. Ignition time

c. Burn duration or incremental velocity

d. Attitude at ignition

e. REFSMMAT or IMU gimbal angles

Outputs required:

The outputs required for this processor are displayed on the FDO detailed maneuver table shown in Figure 3.

4.5.3 Maneuver evaluation. - This processor computes a maneuver which is equivalent to the maneuver actually performed. A preburn vector is propagated to the time of the postburn vector; both vectors are then propagated back to the maneuver ignition time. At this point, the actual spacecraft attitude and external ΔV components are computed.

Inputs required:

a. Preburn state vector

b. Postburn state vector

c. Time of ignition

d. REFSMMAT or IMU gimbal angles

e. Roll angle at ignition

Outputs required:

The outputs required for this processor are displayed on the maneuver evaluation summary sheet shown in Figure 8.

4.5.4 Apollo Real-Time Rendezvous Support Program. - The Apollo Real-Time Rendezvous Support (ARRS) Program was designed to provide a dual purpose tool for both mission planning and rendezvous mission support in the RTACF. ARRS is composed of a number of processors and routines required to support a rendezvous mission. The processors and routines which will be of concern to the Apollo 7 mission are described below:

a. The general purpose maneuver processor (GPMP) is used to compute impulsive maneuvers at a specified point in an orbit to achieve desired orbital conditions.

b. The two-impulse and terminal phase processor computes a set of two impulsive maneuvers by specifying when they should be performed and by specifying the conditions, such as phase and height offsets, at the final maneuver point.

c. The mission plan table (MPT) processor accepts vectors before and after impulsive maneuvers and computes the required finite burn quantities necessary to achieve the orbit after the maneuver.

d. The relative print routine computes relative quantities, such as range, range rate, and look angles, between two orbiting vehicles.

e. The tracking routine computes the tracking station coverage of a vehicle from its initial vector through all the maneuvers that have been established in the mission plan table.

f. The concentric rendezvous processor computes a rendezvous plan by using concentric flight-plan logic. This processor may be used to compute the second maneuver of the two-impulse rendezvous plan after the first maneuver has been executed by the spacecraft, a capability nonexistent in the two-impulse and terminal phase processor.

g. The conversion routine converts vectors from one coordinate system to a number of other coordinate systems.

h. The trajectory update routine accepts a tracking vector, executes the maneuvers in the MPT, and recomputes tracking station coverage.

The inputs required by the ARRS program are described in the ARRS input manual given in Reference 8.

4.6 Command Load Processors

The RTACF possesses the capability of generating command loads in octal with the proper scaling and format to be directly uplinked to the CMC. The capability also exists for receiving certain downlinked quantities from the CMC and converting them to the appropriate engineering units. A program was developed to perform these special conversions as well as any number of general conversions from engineering units to octal or vice versa. In addition, a processor was developed to generate a CMC or S-IVB state vector update at a given time from a RTCC tracking vector.

4.6.1 Command Formatting and Conversion Program. - The program contains seven options in which data in engineering units are converted to octal format, or data in octal are converted to engineering units. Six of the options are concerned with up- or down-linked CMC quantities and possess preset formats, scale factors, and octal precisions. The seventh option is for general conversion from engineering units to octal, or vice versa, given the number, scale factor, precision, and any multipliers.

To simplify discussions of the seven options, only conversions in one direction will be considered. It should be noted, however, that each option also contains the capability to convert in the other direction.

1. Navigation vector update: The navigation vector update option converts a state vector in the Besselian coordinate system with the units of feet and feet per second to octal units acceptable to the CMC.

Inputs required:

a. Position components of a state vector in feet

b. Velocity components of a state vector in feet per second

c. Vector time in ground elapsed time

Outputs required:

The outputs required from this option are displayed on the command load navigation update summary sheet shown in Figure 9.

2. Orbital external ΔV: This option converts the target external ΔV components for an orbital maneuver, the maneuver time, and the weight prior to the maneuver to octal units to be uplinked to the CMC.

Inputs required:

a. Maneuver time in ground elapsed time

b. External ΔV components in feet per second

c. Maneuver weight in pounds

Outputs required:

The outputs required from this option are displayed on the orbital external ΔV summary sheet shown in Figure 10.

3. Deorbit external ΔV: This option converts the target deorbit ΔV components, the deorbit ignition time, the weight at deorbit, and the target point to the octal format required for CMC uplink.

Inputs required:

a. Maneuver time in ground elapsed time

b. External ΔV components in feet per second

c. Maneuver weight in pounds

d. Latitude and longitude of the target in degrees

Outputs required:

The outputs required from this option are displayed on the deorbit external ΔV summary sheet shown in Figure 11.

4. REFSMMAT update: This option converts a REFSMMAT to the octal format required for a CMC update.

Inputs required:

Elements of REFSMMAT

Outputs required:

The outputs required from this option are displayed on the REFSMMAT update summary sheet shown in Figure 12.

5. Numeric RTCC restart: This option converts a spacecraft vector in the CMC numeric units to an RTCC vector in engineering units. This conversion routine contains the transformation from the Besselian to the Greenwich inertial coordinate system.

Inputs required:

a. Position components of spacecraft vector in numeric units

b. Velocity components of spacecraft vector in numeric units

c. Vector time in numeric units

Outputs required:

The outputs required from this option are displayed on the command load navigation update summary sheet shown in Figure 9.

6. Alphanumeric RTCC restart: This option is similar to option 5 except that the inputs are in the CMC alphanumeric units.

Inputs required:

Same as option 5 with input in alphanumeric units

Outputs required:

Same as option 5.

7. General octal conversion: This option converts any number from engineering units to octal after specifying the scale factor and the octal precision. The option also has the capability of converting the number from one set of engineering units to another set by specifying a multiplier.

Inputs required:

a. Number to be converted

b. Scale factor

c. Octal precision

d. Multiplier

Outputs required:

The outputs required from this option are displayed on the general octal conversion summary sheet shown in Figure 13.

4.6.2 Navigation vector update. - This GEMMV processor takes an RTCC state vector, propagates it forward to the navigation vector update time, and outputs the state vector at that time in engineering units and either the CMC or S-IVB octal format.

Inputs required:

a. RTCC state vector

b. Time of navigation vector update

c. Spacecraft weight, drag coefficient, and reference area

Outputs required:

The outputs required for this processor will be displayed on either the CMC or S-IVB navigation vector update summary sheet shown in Figures 9 and 14, respectively.

4.7 Optical Sighting Processors

The optical equipment aboard the CM consists of the following instruments: a scanning telescope, a sextant, and a boresight. The scanning telescope and sextant are interconnected with equipment common to the inertial and computer subsystems to form the primary onboard optical navigation subsystem. The scanning telescope is a single line-of-sight, unit power, wide field instrument. It is used for landmark tracking, in conjunction with IMU attitude reference, as an acquisition instrument for the sextant, for backup alignment of the inertial attitude sensors, and as a general viewing instrument. The sextant is a highly accurate, dual line-of-sight, 28-power telescope with a 1.8-degree field of view. It is primarily used to determine star to landmark or horizon angle navigation measurements, to compute star measurements for IMU alignment, and as a high power general viewing instrument.

The boresight is not part of the optical navigation subsystem and has no coupling with either the inertial or computer subsystems. It is used as a general viewing instrument and as a backup for IMU alignment if the optical subsystem should fail. It is a compact, low magnification, 5-degree field-of-view telescope whose null position line of sight is parallel to the spacecraft roll axis.

Two programs were developed for use in the RTACF which satisfy the optical sighting requirements. The Guidance Optical Support Table (GOST) Program was developed to verify any new IMU alignment by computing the star sighting angles for the sextant and boresight and the view-field coordinates of stars with respect to the telescope reticle patterns. The Star Sighting Table (SST) Program determines the time, attitude, and other conditions necessary for sighting specified ground and celestial targets.

4.7.1 Guidance Optical Support Table Program. - The GOST Program is not an integrating program but requires the assistance of an integrating program to generate a state vector, vehicle attitude, and other related parameters at the time of the optical sighting. Therefore, the GOST Program is executed as a postprocessor to the GEMMV general purpose processor which generates the trajectory data needed for initialization.

The program contains three options all of which use the standard GOST summary sheet shown in Figure 15. In the program description which follows, the inputs listed for each program option are those required by the GOST Program with no distinction made between those input directly and those obtained from the GEMMV general purpose processor.

1. IMU alignment: The REFSMMAT corresponding to a new IMU alignment is computed given the identification of the two stars sighted, the sextant shaft and trunnion angles for each star, and the spacecraft attitude at the time of the star sightings.

Inputs required:

a. RTCC state vector

b. Spacecraft attitude in IMU gimbal angles

c. Two star identifications

d. The sextant shaft and trunnion angles of each star

e. Time of star sighting

Outputs required:

REFSMMAT associated with the new IMU alignment.

2. Star finding: The star finding option locates two stars which are in the scanning telescope field of view at a specified spacecraft attitude and IMU alignment. The two stars must satisfy the condition that one star lies on the R-line and the other star lies as close as possible to the M-line of the telescope reticle pattern.

Inputs required:

a. RTCC state vector

b. REFSMMAT

c. IMU gimbal angles at time of sighting

d. Time of star sighting

Outputs required:

a. Two stars in the scanning telescope field of view which satisfy the reticle constraint

b. Telescope shaft and trunnion angles necessary to place the stars in the center of the reticle pattern

c. Telescope reticle location of the two stars

d. Sextant shaft and trunnion angles of the two stars

3. Star location: The star location option determines the location of two stars which are in the CM optical system field of view at the current spacecraft attitude and IMU alignment.

Inputs required:

a. RTCC state vector

b. REFSMMAT

c. IMU gimbal angles

d. Two star identifications

e. Time of star sighting

Outputs required:

The sextant shaft and trunnion angles for each star.

4.7.2 Star Sighting Table Program. - The SST Program was developed to satisfy those optical sighting requirements which are concerned with ground and celestial target sightings. The program computes the time of arrival at a specified line of sight to the target and the spacecraft attitude or sextant orientation required to view the target.

The program requires an ephemeris to generate the target sighting data, but it does not contain the capability of propagating state vectors. Therefore, the GEMMV general purpose processor is used to generate an ephemeris tape and the SST Program is executed as a postprocessor using this tape.

The program contains three options, the outputs of which are displayed on the star sighting table shown in Figure 16. In the description of each option which follows, the inputs listed are the combined inputs required for both the GEMMV general purpose processor and the SST Program option.

1. Ground target sighting: This option computes the required spacecraft attitude for viewing a specified ground target at a given line-of-sight elevation angle with the CM optical system in a fixed configuration.

Inputs required:

a. RTCC state vector

b. REFSMMAT

c. Target identification or location

d. Elevation angle of line of sight to the target

e. Sextant shaft and trunnion angles

Outputs required:

a. IMU gimbal angles

b. Time of arrival at the desired line of sight to the ground target

c. Central angle and time of closest approach

2. Celestial target sighting with fixed sextant configuration: This option computes the required spacecraft attitude and time for viewing a specified celestial target with the sextant in a fixed configuration.

Inputs required:

a. RTCC state vector

b. REFSMMAT

c. Target identification or location

d. Sextant shaft and trunnion angles

Outputs required:

a. IMU gimbal angles

b. Time of arrival at the line of sight

c. Minimum target-earth-spacecraft central angle (closest approach)

d. Time of closest approach

e. Right ascension and declination of the line of sight

f. Earliest subsatellite longitude at which the line of sight does not pass through the earth's atmosphere

3. Celestial target sighting with moveable sextant configuration: This option computes the required sextant shaft and trunnion angles for viewing a celestial target at a fixed spacecraft attitude. The inputs and outputs from this option are identical to those in option 2 with the exception that the spacecraft attitude is input and the sextant shaft and trunnion angles are output.

4.8 Work Schedule Processor

The RTACF work schedule processor was developed to display, in graphical form, those mission and orbit related events which occur in a specified interval of time during the mission. The processor was intended to operate in a real-time environment and to generate a work schedule which would reflect any anomalies or alternate procedures which might develop during the mission.

The work schedule processor is divided into three separate modules. Module I employs any orbit phase GEMMV processor and is used to generate an ephemeris tape which becomes the input to the next module. The ephemeris tape contains all the pertinent orbit and maneuver data in the specified time interval. Module II processes the ephemeris tape and has the capability of generating any of the following data: radar, spacecraft daylight-darkness, spacecraft moon sighting, computed events, landmark sighting, spacecraft star sighting, closest approach, and pointing data. These data are also saved on an interface tape which serves as the input to Module III together with any comments to be included in the work schedule. The processor may be terminated at this point if only the results from Module II are desired. The processor output then consists of the summary sheets which are shown in Figures 17 through 24.

The execution of Module III is performed when the work schedule is desired. The module sorts the information contained on the interface tape and generates a plot tape which is converted to the work schedule format shown in Figure 25.

The data generated in Module II comprise the bulk of the information contained in the work schedule. The eight Module II options are described below, and the inputs required for execution of these options are listed. These inputs are in addition to the following Module I inputs which are used to generate the ephemeris tape.

a. RTCC state vector

b. Spacecraft weight

c. Time interval to be considered.

d. Orbital maneuver timeline

1. Radar: The radar option computes the spacecraft acquisition and loss times, maximum elevation, and slant range for all spacecraft passes over network stations for some specified interval of time.

Inputs required:

a. Network stations desired

b. Minimum acceptable elevation angle

c. Time interval in which radar data is needed

Outputs required:

The output quantities from this option are displayed on the radar summary sheet shown in Figure 17.

2. Daylight-darkness: The daylight-darkness option computes the time and position at which the spacecraft enters and leaves the earth's shadow and the geodetic latitude and longitude of terminator rise and set.

Inputs required:

Time interval in which daylight-darkness data are required.

Outputs required:

The output quantities from this option are displayed on the daylight-darkness summary sheet shown in Figure 18.

3. Moon sighting: The moon sighting option computes the time and position in which the moon is visible from the spacecraft.

Inputs required:

Time interval in which moon sightings are required.

Outputs required:

The output quantities from this option are displayed on the moon sighting summary sheet shown in Figure 19.

4. Computed events: The computed events option computes the time, position, and altitude of apogee and perigee in addition to the time and location of the ascending node and the Cape crossing time.

Inputs required:

Time interval in which computed events are required.

Outputs required:

The output quantities from this option are displayed on the computed events summary sheet shown in Figure 20.

5. Landmark sighting: The landmark sighting option computes the spacecraft acquisition and loss times, maximum elevation, and slant range for all spacecraft passes over the specified landmarks during the requested interval of time.

Inputs required:

a. Landmark number

b. Time interval during which landmark sightings are required

Outputs required:

The output quantities from this option are displayed on the landmark sighting summary sheet shown in Figure 21.

6. Star sighting: The star sighting option determines the time interval during which a star is visible from the spacecraft and the closest approach (minimum star-earth-spacecraft central angle) data associated with the star sighting in the requested time interval. The closest approach data include the time, central angle, and spacecraft attitude at the closest approach.

Inputs required:

a. Star identification

b. Time interval during which star sightings are required

Outputs required:

The output quantities from this option are displayed on the star sighting summary sheet shown in Figure 22.

7. Closest approach: The closest approach option computes the time, position above the earth, and altitude of the spacecraft at its closest approach to a ground target.

Inputs required:

a. Target identification

b. Time interval during which these data are required

Outputs required:

The output quantities from this option are displayed on the closest approach summary sheet shown in Figure 23.

8. Pointing data: The pointing data option computes the spacecraft to target and target to spacecraft look angles as well as the normal radar tracking data. This option has also been designed to satisfy the lunar module (LM) rendezvous radar requirement.

Inputs required:

a. Target identification

b. Time interval during which pointing data are required

c. REFSMMAT

Outputs required:

The output quantities from this option are displayed on the pointing data summary sheet shown in Figure 24.

4.9 CSM Systems Programs

The CSM systems programs were designed to keep a record of the amount of consumables used during the mission and to update the CSM mass properties to reflect this consumption and any CSM reconfigurations. Three programs are employed in the RTACF to generate these data. One program computes the mass properties of the CSM for a specified CSM configuration and can determine the CM trim aerodynamics for entry. The other programs are used to determine the amount of SM RCS propellant remaining by either estimating the amount of propellant used for each maneuver or by computing the amount of propellant remaining in each tank from the measured values of temperature and pressure of the helium used to pressurize the fuel-oxidizer system.

4.9.1 Mass properties and aerodynamics. - The mass properties and aerodynamics program was developed to determine the CM entry aerodynamics in addition to the CSM center of gravity location, moments of inertia, and engine trim angles at any time during the mission. The program is composed of four options from which the CM trim aerodynamics, CM or CSM center of gravity location, mass properties table, and digital autopilot command load can be generated.

1. Aerodynamics: The aerodynamics option computes the trim aerodynamic coefficients of the command module as a function of Mach number.

Inputs required:

a. Weight of present CM configuration

b. X, Y, and Z components of the center of gravity.

Outputs required:

The outputs required are displayed on the aerodynamics update summary sheet shown in Figure 26.

2. Center of gravity: The center of gravity option is used to determine a new vehicle center of gravity location for any desired configuration of CSM. This option also contains the capability of generating CM entry trim aerodynamics for any new configuration of the CM.

Inputs required:

a. Weight, center of gravity position, and moments of inertia of the CM, SM, and modules to be added or subtracted.

b. Total number of modules to be considered.

Outputs required:

The outputs required are displayed on the center of gravity summary sheet shown in Figure 27.

3. Mass properties table: This option generates a table of center of gravity locations, moments of inertia, and engine trim angles as a function of the CSM weight for a specific oxidizer to fuel mixture ratio. A portion of the table (center of gravity position as a function of weight) is output on punched cards in a format acceptable to the RTCC and RTACF trajectory programs.

Inputs required:

a. Dry weight of CM and SM

b. Weight of consumables

c. CM and SM center of gravity locations and moments of inertia.

d. SPS oxidizer to fuel mixture ratio

e. Weight, center of gravity location, and moments of inertia of any items to be considered

Outputs required:

The outputs required are displayed on the mass properties summary sheet shown in Figure 28.

4. Digital autopilot command load: This option computes those mass properties required for uplink to the digital autopilot. The quantities are converted to the proper units and octal format acceptable to the CMC digital autopilot program.

Inputs required:

a. Dry weight of CM and SM

b. Weight of consumables

c. CM and SM center of gravity locations and moments of inertia

d. SPS thrust level

e. Weight, center of gravity location, and moments of inertia of any items to be considered

Outputs required:

The outputs required are displayed on the digital autopilot command load summary sheet shown in Figure 29.

4.9.2 Mass Properties, Reaction Control System, Service Propulsion System (MRS) Program. - The MRS Program is designed to generate a complete RCS propellant budget using premission data supplied for individual maneuver propellant consumption and internally computed mass properties characteristics. During the mission, as propellant is expended and vehicle configuration modified, the RCS portion of the program accepts inputs from the mass properties portion for use in its computations. In addition to mass properties, the RCS portion uses a form of flight timeline which is input by the user and fixed data which are stored in the program. The program is used in real-time mission support to correct the preflight budget in accordance with changes in the basic flight plan or procedure.

The input and output description of the MRS Program is given in the MRS Program description, which is presented in Reference 9.

4.9.3 Pressure, volume, temperature (PVT) equation for SM RCS. - This program determines the amount of SM RCS oxidizer and fuel remaining in each tank and how much of this can be considered useful propellant. From telemetered values of helium temperature and pressure, the program employs the gas equation to determine the volume of helium used to pressurize the fuel-oxidizer system. Once the volume of helium is determined in each tank, the amount of fuel or oxidizer is computed from the known total volume of each tank. The amount of usable propellant is then determined from the oxidizer to fuel mixture ratio being used and the expulsion efficiency of each tank.

Inputs required:

a. Volume of each oxidizer, fuel, and helium tank

b. Volume of all connecting lines

c. Initial pressure and temperature of the helium tank

d. Initial weight of fuel and oxidizer in each tank

e. Helium source pressure and temperature at the time of the PVT calculation

f. Fuel and oxidizer manifold pressure

g. Oxidizer to fuel mixture ratio

h. Weight of oxidizer and fuel remaining in the passive system (primary or auxiliary)

i. Expulsion efficiency of each tank

Outputs required:

The outputs required are displayed on the PVT summary sheet shown in Figure 30.

4.10 Deorbit Processors

There are eight deorbit processors in the RTACF to support the Apollo 7 mission. Two processors were developed to simulate deorbits to primary landing areas while deorbits to contingency landing areas required the use of five processors. Two of these five processors simulate hybrid deorbits employing both the SM and CM RCS thrusters. The eighth is a block data program designed to generate a series of deorbits including CLA's, PLA's, and SM RCS apogee deorbits.

4.10.1 Primary landing area. - The PLA processors were developed to simulate a targeted deorbit into a primary landing area. This area is a region about a predetermined landing point where recovery forces have been stationed or can be made available in a short period of time. The target point in the processors is defined by a latitude and longitude, and the processor iterates on both the deorbit ignition time and the time to reverse the bank angle to achieve the specified target.

1. PLA processor 1: This processor employs an iterative technique to determine the SPS or RCS deorbit ignition time and time to reverse bank to land in a primary landing area. The deorbit maneuver

is determined by the following quantities: an initial attitude, a guidance mode, and a terminating value for the maneuver. The initial vehicle attitude can be input with respect to the local vertical/local horizontal (LVLH) system, or aft-looking line of sight to the horizon. The attitude may also be specified by indicating the initial thrust vector attitude with respect to the LVLH system, or by indicating the IMU gimbal angles and stable member alignment. The spacecraft orientation, during the maneuver, is maintained inertial during the maneuver or by either the SCS or external ΔV guidance mode in which the thrust vector is held inertial. The maneuver thrust may be terminated after a specified value of one of the following conditions has been satisfied: an incremental velocity change, external ΔV onboard targets, a burn duration, or a velocity and flight-path angle constraint at entry interface.

There are two entry profiles that can be used to achieve the target latitude and longitude. One profile maintains a constant bank angle from entry interface to the computed time to reverse bank followed by the negative of the initial bank angle to drogue chute deployment. The other profile employs the following sequence: a constant lift vector orientation to a specified g-level, a constant bank angle to the time to reverse bank, and the negative of the previous bank angle to drogue chute deployment.

Inputs required:

a. RTCC state vector

b. CSM weight

c. CM weight at entry

d. REFSMMAT or IMU gimbal angles

e. Deorbit SPS or RCS maneuver data

f. Ignition time (initial guess)

g. Entry profile

h. Latitude and longitude of target

Outputs required:

a. Deorbit ignition time

b. Total ΔV along the X-body axis less tailoff

c. External ΔV components at ignition

d. Duration of the burn

e. IMU gimbal angles at ignition (if not input)

f. Latitude, longitude, altitude, and true anomaly at ignition

g. Thrust vector pitch at ignition

h. Velocity, flight-path angle, and ground elapsed time at 400,000 feet

i. Time from retrofire to 400,000 feet

j. Time from retrofire to a specified number of g's

k. Blackout data

l. Maximum g-level

m. Time from retrofire to drogue chute deployment

n. Time from retrofire to main chute deployment

o. Latitude, longitude, and ground elapsed time of landing

p. Time from retrofire to the time of reverse bank

Optional outputs:

Footprint data

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 2.

2. PLA processor 2: This processor is essentially the same as the PLA processor 1 with the additional capability of simulating the S-IVB/CSM separation maneuver. The separation maneuver can be simulated at either a specified time or at a fixed time prior to the deorbit maneuver. The fixed-time separation, using the SPS or RCS thrusters, is computed based on the desired time of ignition, the spacecraft or thrust vector attitude, incremental velocity to be added, and either the MTVC or SCS guidance mode. Simulating the separation at a fixed-time interval before the deorbit maneuver requires the same inputs as the fixed-time separation except for the ignition time. Instead of specifying the time of ignition, a constant Δt between the separation and deorbit maneuvers is input. The processor then employs an iterative technique to determine the ignition time of the separation maneuver and, hence, the deorbit ignition time.

Inputs required:

The inputs required are identical to PLA processor 1 with the following additions:

i. S-IVB/CSM weight before separation

j. SPS or RCS separation maneuver data

Outputs required:

The outputs required are identical to PLA processor 1 with the following additions:

q. Separation maneuver time

r. Time from separation to retrofire

Optional outputs:

Footprint data

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 2.

4.10.2 Contingency landing area. - In the event of an inability to deorbit into a primary landing area, any suitable area along the groundtrack can be designated as a contingency landing area. The CLA processors can simulate a deorbit to a contingency landing area by targeting only for a longitude. The deorbit ignition time is varied to achieve the target longitude using a fixed entry profile.

1. CLA processor 1: This CLA processor employs an iterative technique to determine the SPS or RCS deorbit ignition time to hit a target longitude in a contingency landing area using a fixed entry profile. The deorbit maneuver, like PLA processor 1 described above, is determined by an initial attitude, a guidance mode, and a terminating value for the maneuver. The entry profile used may be any one of the following:

a. A constant bank angle maintained from entry interface to drogue chute deployment

b. A lift vector orientation from entry interface to a specified g-level followed by a constant bank angle to drogue chute deployment

c. A rolling entry profile either maintaining a constant bank angle from entry interface to 300,000 feet, or a constant lift vector orientation from entry interface to a specified g-level, followed by a rolling entry to 75,000 feet and then a full lift attitude to drogue chute deployment

Inputs required:

The inputs required are identical to the PLA processor 1 with the following exception:

h. Longitude of target

Outputs required:

The outputs required are identical to the PLA processor 1 except that the time from retrofire to the time to reverse bank angle does not apply.

Optional outputs:

Footprint data

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 2.

2. CLA processor 2: This processor is essentially the same as the CLA processor 1 described above, with the additional capability of simulating the S-IVB/CSM separation. The separation can occur at either a fixed time or fixed Δt before the maneuver as described in the PLA processor 2.

Inputs required:

The inputs required are identical to the PLA processor 1 with the following additions:

h. Longitude of target

i. S-IVB/CSM weight before separation

j. SPS or RCS separation maneuver data

Outputs required:

The outputs required are identical to the PLA processor 1 with the following additions:

p. Separation maneuver time

q. Time from separation to retrofire

Optional outputs:

Footprint data

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 2.

3. CLA processor 3: This processor employs an iterative technique to determine the CM RCS ignition time required to deorbit the CM to a contingency landing area. The maneuver can be performed in either the SCS or MTVC guidance mode with the ignition time determined by the incremental velocity to be added, initial pitch attitude, and the target longitude. The CM entry profile will be defined by the following sequence: a constant lift vector orientation from entry interface to a specified g-level

will be flown, followed by a constant roll rate to an altitude of 75,000 feet, and then full lift to drogue chute deployment.

Inputs required:

Same as for CLA processor 1

Outputs required:

Same as for CLA processor 1

Optional outputs:

Footprint data

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 2.

4.10.3 Hybrid deorbit. - The hybrid deorbit processors simulate a deorbit using the SM and CM RCS thrusters in the event of an inability to use the SPS engine. The engine may be inoperative because of an inability to start it, or due to a failure to separate the CSM from the S-IVB. Two processors have been developed to simulate these conditions.

1. Hybrid deorbit processor 1: If a situation arises whereby the SPS engine cannot be used to deorbit the CSM and there is insufficient SM RCS propellant to perform an RCS deorbit, a hybrid deorbit will be performed. The hybrid deorbit consists of a SM RCS burn, CM/SM separation, and a CM RCS burn to achieve a deorbit to a specified contingency landing area.

This hybrid deorbit processor employs an iterative scheme to determine the RCS ignition times in order to achieve a target longitude. The SM RCS burn duration is based on a fixed incremental velocity to be realized by the SM RCS burn. The spacecraft maintains an inertial attitude which is specified in the LVLH plane at the centroid of the hybrid deorbit. A 60-second coast between the SM and CM RCS burns allows time to perform the CM/SM separation and the reorientation of the CM for the CM RCS maneuver. The CM attitude is maintained such that the thrust vector alignment is the same as the SM RCS thrust vector alignment. During entry a specified lift vector orientation is maintained to a fixed g-level, followed by a specified lift entry profile to drogue chute deployment.

This processor can also be used to evaluate the actual hybrid deorbit performed. The SM RCS maneuver is assumed to have been performed nominally while the CM RCS maneuver is defined by the actual incremental velocities achieved. The processor remains essentially unchanged except that the SM RCS ignition time and actual velocity increments are input and the landing point is determined.

Inputs required:

Same as CLA processor 2

Outputs required:

Same as CLA processor 2

Optional outputs:

Footprint data

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 2.

2. Hybrid deorbit processor 2: A hybrid deorbit will be performed in the event of a failure to deploy the SLA panels and a resultant inability to separate the CSM from the S-IVB. The deorbit will consist of a SM RCS burn, the CM separation from the SM/S-IVB, and a CM RCS burn to achieve a landing in a contingency landing area.

This Hybrid deorbit processor uses an iterative technique to determine the SM ignition time in order to achieve a target longitude given the SM and CM incremental velocities to be added and the increment of time needed to perform a separation. The SM maneuver is performed in the S-IVB orbit rate mode which maintains a constant pitch attitude to the local horizontal. After the SM RCS maneuver, the CM separates from the SM/S-IVB and assumes a thrust vector attitude which is specified in the LVLH plane at the centroid of the hybrid deorbit. The CM RCS maneuver is performed by maintaining the thrust vector attitude inertial. The entry profile consists of a lift vector orientation to a specified g-level followed by a specified lift vector orientation to drogue chute deployment.

Inputs required:

Same as CLA processor 2

Outputs required:

Same as CLA processor 2

Optional outputs:

Footprint data

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 2.

4.10.4 Block data. - Block data nominally consist of a series of deorbit maneuvers which can be executed at discrete times during a period of six revolutions to deorbit the CSM in the event of a contingency requiring rapid mission termination. The data is generated and sent to the crew in blocks of six revolutions with three types of deorbits per revolution: an SPS deorbit to a primary landing area, an SPS deorbit to a contingency landing area, and a SM RCS apogee deorbit.

Apollo Block Data Program (ABDP): The ABDP has the capability of performing any number of deorbit maneuvers with or without orbital maneuvers. This program uses an iterative scheme to determine the deorbit ignition time to achieve the target landing point. An orbital maneuver can be performed by specifying the ignition time, the spacecraft attitude, the guidance mode to be used, and an incremental velocity or external ΔV components. The deorbit maneuver is nominally specified by a velocity and flight-path angle constraint at entry interface, spacecraft attitude with respect to the aft-looking line of sight to the horizon, the SCS guidance mode, and a PLA or CLA entry profile. The inputs required to execute the ABDP and the output data and summary sheets, which can be generated, are described in the "Apollo Block Data Program User's Manual" (Reference 10).

4.11 Guided Entry and Backup Guidance Quantities Processors

Guided entries are those whose steering commands are controlled by the entry logic in the onboard computer. As a backup capability to the guidance system, the commander monitors the entry using the entry monitor system display and the flight director attitude indicator. He may take control of the entry at any point and manually fly the CM to touchdown using ground computed backup guidance quantities.

4.11.1 Apollo Reentry Simulation (ARS) Program. - The ARS program takes a state vector at 425,000 feet and computes the necessary entry profile to achieve a specified target latitude and longitude. The state vector is generated by one of the GEMMV deorbit maneuver processors which can either simulate the deorbit maneuver, given a preburn state vector or propagate a postburn state vector to 425,000 feet. At this point the necessary data are saved and then used by ARS to compute the guided entry or necessary backup guidance quantities. There are six entry steering modes in the ARS program. A description of these steering modes is presented below:

1. Automatic guidance and navigation control: In this steering mode the ARS program uses the CMC entry logic to compute the entry steering commands and to simulate the entry trajectory required to achieve the target landing point.

2. Open loop followed by guidance and navigation control: In this entry mode an initial bank angle is maintained from 400,000 feet to a specified g-level at which time the CM is rolled to a second bank angle, designated as the backup bank angle. This attitude is maintained until a second g-level is reached. From this time until drogue chute deployment the ARS program uses the guidance and navigation control logic to compute the steering commands necessary to achieve the target landing point. This steering mode requires the input of an initial and backup bank angle and two g-levels.

3. Bank-reverse-bank: In this entry mode, which is used to compute backup guidance quantities, an initial bank angle is maintained from 400,000 feet to a specified g-level. It is then followed by a backup bank

angle to a computed time to reverse bank and then the reverse bank angle is flown to drogue chute deployment. In this steering mode the initial bank angle and g-level are input and the backup bank angle and time to reverse bank are computed by the ARS program.

4. Combined bank-reverse-bank and guidance and navigation control: This entry mode is the same as that described in the second steering mode with the exception that the program computes the backup bank angle. The inputs consist of the initial bank angle and the two g-levels.

5. Rolling: In this entry an initial bank angle is maintained from 400, 000 feet to a specified g-level followed by a constant roll rate to drogue chute deployment. This mode requires the input of the initial bank angle, g-level, and roll rate.

6. Open loop: This entry can either be a bank-reverse-bank as described in the third steering mode or a constant bank angle entry from 400, 000 feet to drogue chute deployment. The bank-reverse-bank option of this steering mode requires the input of the initial and backup bank angles, the g-level, and the time to reverse bank. A constant bank angle entry can be specified by inputing the value of the bank angle to be used as the initial bank angle and inputing the g-level and time to reverse bank as large values.

Inputs required:

The steering mode and necessary entry quantities.

Outputs required:

The outputs required for this processor are displayed on the ARS summary sheet shown in Figure 31.

Table IV. Data Available from Off-Line Computer Output
Ground elapsed time (hr:min:sec)
Phase time (sec)
Greenwich mean time (hr:min:sec)
Revolution number
X, Y, Z Greenwich and Besselian position coordinates (ft)
$\dot{X}$, $\dot{Y}$, $\dot{Z}$ Greenwich and Besselian velocity coordinates (ft/sec)
Geodetic latitude (deg)
Longitude (deg)
Altitude above oblate earth (ft and n mi)
Altitude above spherical earth (n mi)
Inertial velocity (ft/sec)
Inertial flight-path angle (deg)
Inertial azimuth (deg)
Velocity relative to a rotating earth (ft/sec)
Flight-path angle relative to a rotating earth (deg)
Azimuth relative to a rotating earth (deg)
Aerodynamic velocity (ft/sec)
Aerodynamic flight-path angle (deg)
Aerodynamic azimuth (deg)
True anomaly (deg)
Semimajor axis (ft)
Eccentricity
Inclination (deg)
Argument of perigee (deg)
Orbital period (min)

Table IV. Data Available from Off-Line Computer Output (Continued)

Perigee and apogee altitudes (n mi)
Geocentric body roll, pitch, and yaw angles (deg)
Angle of attack (deg)
Angle of sideslip (deg)
Mach number
Vehicle weight (lb)
IMU gimbal angles (deg)
Body X, Y, and Z-axis load factors (g)
Total load factor (g)
REFSMMAT
Phase change in velocity along body X, Y, and Z-axis (ft/sec)
Total change in velocity along body X-axis (ft/sec)
Total velocity produced by a burn (ft/sec)
X, Y, and Z location of the center of gravity (inches)
Range over spherical earth (n mi)
Thrust during burn phases (lb)
Pitch and yaw trim angles (deg)
External ΔV components in X, Y, and Z at ignition and the centroid (ft/sec)
ΔV's in RCS control axis system

MODE I LAUNCH ABORT DATA

TA SEC	RANGE FT	RANGE NM	BEAR[1] E/N	RAN E FT	RAN N FT	GET IP SEC	LAT-D DEG	LONG DEG	MAX ALT FT-MSL	DD ALT FT	MD ALT FT
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX

[1] MINUS SIGN INDICATES WEST OF NORTH (W/N)

Figure 1. Mode I Launch Abort Summary Sheet

VECTOR IDENTIFICATION

EVENT TIMES	RET			GMT				GET			LATITUDE		LONGITUDE	
	HR	MIN	SEC	DAY	HR	MIN	SEC	HR	MIN	SEC	DEG	MIN	DEG	MIN
RETROFIRE	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
BURN TERMINATION	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
ENTRY INTERFACE	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
BEGIN BLACKOUT, S BAND	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
BEGIN BLACKOUT, VHF	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
.20 G'S	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
REVERSE BANK	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
END BLACKOUT, VHF	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
END BLACKOUT, S BAND	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
DROGUE DEPLOY	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
MAIN DEPLOY	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
LANDING	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX

BURN QUANTITIES

DELTA VELOCITY(X)		XXXXX		FT/SEC
DELTA VELOCITY(C)		XXXXX		FT/SEC
DELTA VELOCITY(T)		XXXXX		FT/SEC
DELTA TIME		XXXXX		SEC
WEIGHT		XXXXX		POUNDS
TRUE ANOMALY		XXXXX		DEG
THRUST PITCH		XXXXX		DEG
LATITUDE	XX	DEG	XX	MIN
LONGITUDE	XXX	DEG	XX	MIN
ALTITUDE		XXXXX		N MI
EXTERNAL DELTA V				
DX		XXXXX		FT/SEC
DY		XXXXX		FT/SEC
DZ		XXXXX		FT/SEC

GIMBAL ANGLES	OUTER (ROLL)	INNER (PITCH)	MIDDLE (YAW)
BURN	XXX	XXX	XXX
ENTRY INTERFACE	XXX	XXX	XXX
REVERSE BANK, BEGIN	XXX	XXX	XXX
REVERSE BANK, END	XXX	XXX	XXX

REENTRY QUANTITIES

VELOCITY(I) EI	XXXXX	FT/SEC
GAMMA(I) EI	XXXXX	DEGREES
WEIGHT	XXXXX	POUNDS
BANK ANGLE	XXXXX	DEG
LIFT	XXXXX	
MAX G'S	XXXXX	

LANDING POINTS

LATITUDE	TARGET	XX	DEG	XX	MIN
LONGITUDE	TARGET	XXX	DEG	XX	MIN
LATITUDE	IMPACT	XX	DEG	XX	MIN
LONGITUDE	IMPACT	XXX	DEG	XX	MIN

Figure 2. Standard Summary Sheet

FDO DETAILED MANEUVER TABLE

CSM STA ID					REF			LH STA ID CODE			
					EXECUTE						
DV-C	XXX	VG-X	XXX	P-I	XXX	V-F	XXX	Y-H	XXX		
DV-M	XXX	VG-Y	XXX	Y-M	XXX	V-S	XXX	P-H	XXX		
DT-U	XXX	VG-Z	XXX	R-O	XXX	V-D	XXX	R-H	XXX	WT-I	XXXXX
DT-B	XXX	GETI	XXX XX XX			DV-TO	XXX	DT-TO	XXX	WM-F	XXXXX

RESULTANT

H-BI	XXX	GET-HA	XXX XX XX	GET-HP	XXX XX XX	D-H	XX
LAT-BI	XX XX XX	H-A	XXXX	H-P	XXXX	THETA	XX
LON-BI	XXX XX XX	LAT-A	XX XX XX	LAT-P	XX XX XX	D-THETA	XX
TA-BI	XXX	LON-A	XXX XX XX	LON-P	XXX XX XX	DELTA	XX
GET-AN	XX XX XX	LON-AN	XXX XX XX	E	XXX	I	XXXX

THRUST

TRIM-P XX TRIM-Y XX

TARGET

EXT-DV		AGS	
GET I	XXX XX XX	GET I	XXX XX XX
V-X	XXX	V-X	XXX
V-Y	XXX	V-Y	XXX
V-Z	XXX	V-Z	XXX

Figure 3. FDO Detailed Maneuver Table

FDO ORBIT DIGITALS

GET	XXX:XX:XX	
REV	XXX	
STA ID		
H	XXXX	
VI	XXXX	
GAM I	XXXX	
A	XXXX	
E	XXXX	
I	XXXX	
HA	XXXX	
LAT A	XX:XX	N/S
LON A	XXXX:XX	E/W
GET A	XXXX:XX:XX	
HP	XXXX	
LAT P	XX:XX	N/S
LON P	XXXX:XX	E/W
GET P	XXXX:XX:XX	

LON PP	XXX:XX	E/W
LAT PP	XX:XX	N/S
GET CC	XXX:XX:XX	
T-A	XXXX	
LON AN	XXX:XX	E/W

Figure 4. FDO Orbit Digital Summary Sheet

RELATIVE MOTION DIGITALS

CSM STA ID　　　　VEH　　　　LM STA ID

REFSMMAT　　　　X AXIS AT TARGET

	GET		R	RDOT	AZH	ELH	X/P	Z/Y	Y/R
XXX	XX	XX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
							XXX	XXX	XXX
XXX	XX	XX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
							XXX	XXX	XXX
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Figure 5. Relative Motion Summary Sheet

RADIATION EVALUATION

LONG DEG	LAT DEG	ALT NM	B GAUSS	L ER	DAY	HR	MIN	SEC	DOSE RATE (REM/HR) LM SKIN	DOSE RATE (REM/HR) CM SKIN	DOSE RATE (REM/HR) CM DEPTH	CUMULATIVE DOSE (REM) LM SKIN	CUMULATIVE DOSE (REM) CM SKIN	CUMULATIVE DOSE (REM) CM DEPTH
XXXX	XXXX	XXXX	XXXX	XXXX	XX	XX	XX	XX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX
XXXX	XXXX	XXXX	XXXX	XXXX	XX	XX	XX	XX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX
XXXX	XXXX	XXXX	XXXX	XXXX	XX	XX	XX	XX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX
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Figure 6. Radiation Evaluation Summary Sheet

GROUND TRACK

REV NO.	GET HRS	GET MIN	GET SEC	GMT DAY	GMT HRS	GMT MIN	GMT SEC	LAT DEG	LAT MIN	LONG DEG	LONG MIN	ALT NO.	AZIMUTH DEGREES
XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXXX	XXXXX
XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXXX	XXXXX
XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXXX	XXXXX
XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXXX	XXXXX
XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXXX	XXXXX
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Figure 7. Ground Track Summary Sheet

MANEUVER EVALUATION OUTPUT

DELTA V	TOTAL=	XXXXX	
	DELTA VX=	XXXXX	
	DELTA VY=	XXXXX	
	DELTA VZ=	XXXXX	
PITCH=	XXXXX	YAW=	XXXXX
GIMBAL ANGLES			
OUTER=	XXXXX		
INNER=	XXXXX		
MIDDLE=	XXXXX		

Figure 8. Maneuver Evaluation Summary Sheet

CSM NAV UPDATE TO CMC

LOAD NO　　GET GEN　　SITES

STA ID　　GMT ID

OID	FCT	DSKY V71	VECTOR
1	INDEX		
2	ADD		
3	VID		
4	X	XXXXX	XXXXXXX
5	X	XXXXX	
6	Y	XXXXX	XXXXXXX
7	Y	XXXXX	
10	Z	XXXXX	XXXXXXX
11	Z	XXXXX	
12	X-DOT	XXXXX	XXXXXXX
13	X-DOT	XXXXX	
14	Y-DOT	XXXXX	XXXXXXX
15	Y-DOT	XXXXX	
16	Z-DOT	XXXXX	XXXXXXX
17	Z-DOT	XXXXX	
20	T	XXXXX	XXX XX XX
21	T	XXXXX	

Figure 9. Command Load Navigation Update Summary Sheet

CMC EXT DELTA-V UPDATE

LOAD NO SITES

STA ID GMT ID

GET GEN MAN CODE

OID	FCT	DSKY V71	DECIMAL
1	INDEX		
2	ADD		
3	TIGN	XXXXX	XXX XX XX
4	TIGN	XXXXX	
5	VGX	XXXXX	XXXXX
6	VGX	XXXXX	
7	VGY	XXXXX	XXXXX
10	VGY	XXXXX	
11	VGZ	XXXXX	XXXXX
12	VGZ	XXXXX	
13	MAN WT	XXXXX	XXXXX

Figure 10. Orbital External ΔV Summary Sheet

CMC RETROFIRE EXTERNAL DELTA-V UPDATE

TTI	STA ID		TYPE		LOAD NO	
CT	GET GEN		SITES PRI		B/U	
THRUSTER		OID	FCT	DSKY V71	DECIMAL	
AREA		1	INDEX			
		2	ADD			
GMTI		3	LAT	XXXXX	XXXXX	DEG
GETI		4	LAT	XXXXX		
		5	LONG	XXXXX	XXXXX	DEG
BT		6	LONG	XXXXX		
V(C)		7	TIGN	XXXXX	XXX XX XX	
		10	TIGN	XXXXX		
R(O)		11	VGX	XXXXX	XXXXX	FPS
P(I)		12	VGX	XXXXX		
Y(M)		13	VGY	XXXXX	XXXXX	FPS
		14	VGY	XXXXX		
H(P)		15	VGZ	XXXXX	XXXXX	FPS
RT400K		16	VGZ	XXXXX		
RETRB		17	WT	XXXXX	XXXXX	LBS

Figure 11. Deorbit External ΔV Summary Sheet

REFSMMAT UPDATE FORMAT

MAT OID	FCT	DSKY	DECIMAL
01	INDEX		
02	ADD		
03	XIXE	XXXXX	XXXXXXX
04	XIXE	XXXXX	
05	XIYE	XXXXX	XXXXXXX
06	XIYE	XXXXX	
07	XIZE	XXXXX	XXXXXXX
10	XIZE	XXXXX	
11	YIXE	XXXXX	XXXXXXX
12	YIXE	XXXXX	
13	YIYE	XXXXX	XXXXXXX
14	YIYE	XXXXX	
15	YIZE	XXXXX	XXXXXXX
16	YIZE	XXXXX	
17	ZIXE	XXXXX	XXXXXXX
20	ZIXE	XXXXX	
21	ZIYE	XXXXX	XXXXXXX
22	ZIYE	XXXXX	
23	ZIZE	XXXXX	XXXXXXX
24	ZIZE	XXXXX	

Figure 12. REFSMMAT Update Summary Sheet

ENGINEERING UNITS/OCTAL CONVERSION

ENGINEERING NUMBER	OCTAL NUMBER	POWER OF TWO	PRECISION	SCALE FACTOR
XXXXXX	XXXXX XXXXX	XX	XX	XXXXX

Figure 13. General Octal Conversion Summary Sheet

SATURN COMMAND LOAD

NAVIGATION UPDATE

LOAD NO XXXX GETSV XX XX XX

FCT	ENGLISH	MATRIX	OCTAL LOAD DATA				
			11	12	13	14	15
Z DOT	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
			21	22	23	24	25
X DOT	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
			31	32	33	34	35
Y DOT	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
			41	42	43	44	45
Z POS	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
			51	52	53	54	55
X POS	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
			61	62	63	64	65
Y POS	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
			71	72	73	74	75
TIME	XX XX XX	XXXXX	XXX	XXX	XXX	XXX	XXX

Figure 14. S-IVB Navigation Vector Update Summary Sheet

GETAC XXX+XX+XX PITCH XXXXX YAW XXXXX ROLL XXXXX

	GIVEN VALUES		SMOOTHED VALUES	
SEXTANT	SHAFT	TRUNNION	SHAFT	TRUNNION
S1 XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
S2 XXXXX	XXXXX	XXXXX	XXXXX	XXXXX

SCANNING TELESCOPE

STAR				M	R	LAT	LONG
1	XX	SHAFT	XXX	XX	XX	XXX	XXX
2	XX	TRUNNION	XXX	XX	XX	XXX	XXX

BORESIGHT

STAR	RHO	THETA	LAT	LONG
S1 XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
S2 XXXXX	XXXXX	XXXXX	XXXXX	XXXXX

MATRIX

XIXE	XXXXX	XIYE	XXXXX	XIZE	XXXXX
YIXE	XXXXX	YIYE	XXXXX	YIZE	XXXXX
ZIXE	XXXXX	ZIYE	XXXXX	ZIZE	XXXXX

Figure 15. Standard GOST Summary Sheet

STAR SIGHTING TABLE

TGT ID		R-O	XXX		
TGT DEC	XX:XX:XX	P-I	XXX	LOS DEC	XX:XX:XX
TGT RT ASC	XXX:XX:XX	Y-M	XXX	LOS RT ASC	XXX:XX:XX
GND. PT DATA		SFT	XXX	REV	XXX
		TRN	XXX	LAT-LOS	XXXXX N/S
LAT	XXXXX N/S			GET-LOS	XXX:XX:XX
LONG	XXXXX E/W				
ALT	XXXXX			W-D	XXXXX
ELV	XXXXX			GETCA	XXX:XX:XX

REFSMMAT

XIX	XXXXXX	YIX	XXXXXX	ZIX	XXXXXX
XIY	XXXXXX	YIY	XXXXXX	ZIY	XXXXXX
YIZ	XXXXXX	YIZ	XXXXXX	ZIZ	XXXXXX
THETA-X	XXXXX	THETA-Y	XXXXX	THETA-Z	XXXXX

Figure 16. Star Sighting Table

RADAR TRACKING

VEHICLE 1

			ACQUISITION							LOSS OF SIGNAL							DELTA						
STATION		REV	GET			GMT				GET			GMT				T			MAX ELEV	ACQ AZIMUTH	ACQ RANGE	MIN RANGE
CODE		NO.	HR	MIN	SEC	DAY	HR	MIN	SEC	HR	MIN	SEC	DAY	HR	MIN	SEC	HR	MIN	SEC	DEGREES	DEGREES	N.MI.	N.MI.
HTV	R	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
CAL	R V	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
GDS	RTVC	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
GYM	RTVC	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
WHS	C	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
TEX	RTVC	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
PAT	R	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
MLA	R	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
GBI	RTVC	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
ANT	RTVC	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
BDA	RTVC	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
VAN	R	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
CYI	RTVC	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
ASC	R V	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
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Figure 17. Radar Summary Sheet

SPACECRAFT DAYLIGHT-DARKNESS

VEHICLE 1

	GET			GMT				GEODETIC LATITUDE		LONGITUDE		ALTITUDE	SUN AZIM ANGLE	LV/LH	
	HR	MIN	SEC	DAY	HR	MIN	SEC	DEG	MIN	DEG	MIN	N. MI.	DEGREES	PITCH DEG	YAW DEG
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX			
TERMINATOR SET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX			
SUNSET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX	XXX	XXX	XXX
SUNRISE	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX	XXX	XXX	XXX
TERMINATOR RISE	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX			
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX			
TERMINATOR SET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX			
SUNSET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX	XXX	XXX	XXX
SUNRISE	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX	XXX	XXX	XXX
TERMINATOR RISE	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX			
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX			
TERMINATOR SET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX			
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Figure 18. Daylight-Darkness Summary Sheet

SPACECRAFT MOON SIGHTINGS

VEHICLE 1

	GET			GMT				GEODETIC LATITUDE		LONGITUDE		ALTITUDE
	HR	MIN	SEC	DAY	HR	MIN	SEC	DEG	MIN	DEG	MIN	N. MI.
MOONSET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX
MOONRISE	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX
MOONSET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX
MOONRISE	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX
MOONSET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX
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Figure 19. Moon Sighting Summary Sheet

COMPUTED EVENTS

VEHICLE 1

	GET			GMT				ALTITUDE	GEODETIC LATITUDE		LONGITUDE		RIGHT ASCENSION		INCLINATION
	HR	MIN	SEC	DAY	HR	MIN	SEC	N. MI.	DEG	MIN	DEG	MIN	HR	MIN	DEGREES
ASC NODE	XXX	XX	XX	XX	XX	XX	XX	XXXX	XX	XX	XXX	XX	XX	XX	XXX
PERIGEE	XXX	XX	XX	XX	XX	XX	XX	XXXX	XX	XX	XXX	XX	XX	XX	XXX
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XX	XXXX	XX	XX	XXX	XX			
APOGEE	XXX	XX	XX	XX	XX	XX	XX	XXXX	XX	XX	XXX	XX			
ALTITUDE	XXX	XX	XX	XX	XX	XX	XX	XXXX	XX	XX	XXX	XX			
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Figure 20. Computed Events Summary Sheet

LANDMARK SIGHTINGS

VEHICLE 1

LANDMARK	REV	ACQUISITION GET			ACQUISITION GMT				LOSS OF LANDMARK GET			LOSS OF LANDMARK GMT				DELTA T			ACQ AZ	ACQ GRND RANGE	MIN GRND RANGE	MAX ELEV	GET OF MAX ELEV			ALT OF MAX ELEV
NUMBER	NO.	H	M	S	D	H	M	S	H	M	S	D	H	M	S	H	M	S	DEG	N. MI.	N. MI.	DEG	H	M	S	N. MI.
XXX	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	X	XX	XX	XXX	XXXXX	XXXXX	XX	XXX	XX	XX	XXXXX
XXX	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	X	XX	XX	XXX	XXXXX	XXXXX	XX	XXX	XX	XX	XXXXX
XXX	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	X	XX	XX	XXX	XXXXX	XXXXX	XX	XXX	XX	XX	XXXXX
XXX	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	X	XX	XX	XXX	XXXXX	XXXXX	XX	XXX	XX	XX	XXXXX
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Figure 21. Landmark Sighting Summary Sheet

STAR SIGHTINGS

VEHICLE 1

	GET			RIGHT ASCENSION			DECLINATION				CENTRAL ANGLE		ALTITUDE	AZIMUTH
	HR	MIN	SEC	HR	MIN	SEC	DEG	MIN	SEC	MAGNITUDE	DEG	MIN	N. MI.	DEGREES
STAR XX RISE	XXX	XX	XX	XX	XX	XX	XXX	XX	XX	XXXXX	XXX	XX	XXXXX	XXX
STAR XX SET	XXX	XX	XX	XX	XX	XX	XXX	XX	XX	XXXXX	XXX	XX	XXXXX	XXX
STAR XX RISE	XXX	XX	XX	XX	XX	XX	XXX	XX	XX	XXXXX	XXX	XX	XXXXX	XXX
STAR XX SET	XXX	XX	XX	XX	XX	XX	XXX	XX	XX	XXXXX	XXX	XX	XXXXX	XXX
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Figure 22. Star Sighting Summary Sheet

CLOSEST APPROACH

VEHICLE 1

TARGET ACR LAND

LATITUDE DEGREES XXXXX

LONGITUDE DEGREES XXXXX

ALTITUDE FEET XXXXX

		GMT			GET					GIMBAL ANGLES			LV/LH	
REV	DAY	HR	MIN	SEC	HR	MIN	SEC	GROUND RANGE N. MI.	ALTITUDE N. MI.	PITCH (I) DEG	YAW (M) DEG	ROLL (0) DEG	PITCH DEG	YAW DEG
1	XX	XX	XX	XX	XXX	XX	XX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
2	XX	XX	XX	XX	XXX	XX	XX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
3	XX	XX	XX	XX	XXX	XX	XX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
4	XX	XX	XX	XX	XXX	XX	XX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
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Figure 23. Closest Approach Summary Sheet

POINTING DATA

VEHICLE 1

TARGET XXXXX   LATITUDE XXXXX DEG   LONGITUDE XXXXX DEG   ALTITUDE XXXXX N.MI.

REFSMMAT

```
                    * * * * * * * * * * * * * * * * * * * * * * * * *
                    *   X   XXXXX   XXXXX   XXXXX   X   *
REVOLUTION XXX      *   Y   XXXXX   XXXXX   XXXXX   Y   *
                    *   Z   XXXXX   XXXXX   XXXXX   Z   *
                    * * * * * * * * * * * * * * * * * * * * * * * * *
```

	GET HR:MIN:SEC	GMT DAY:HR:MIN:SEC	ELAPSED TIME MIN:SEC	MAX ELEV ANGLE DEG	MINIMUM RANGE N.MI.	ALT MIN RANGE N.MI.
ACQ	XXX XX XX	XX XX XX XX	XX : XX	XX : XX	XXXXX	XXXXX
LOSS	XXX XX XX	XX XX XX XX				

										LVLH	
GET HR:MIN:SEC	GMT DAY:HR:MIN:SEC	RANGE N.MI.	R-DOT FT/SEC	R-ACC FT/SEC2	AZIMUTH DEG	ELEVATION DEG	ROLL(0) DEG	PITCH(I) DEG	YAW(M) DEG	PITCH DEG	YAW DEG
XXX XX XX	XX XX XX XX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
XXX XX XX	XX XX XX XX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
XXX XX XX	XX XX XX XX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX

Figure 24. Pointing Data Summary Sheet

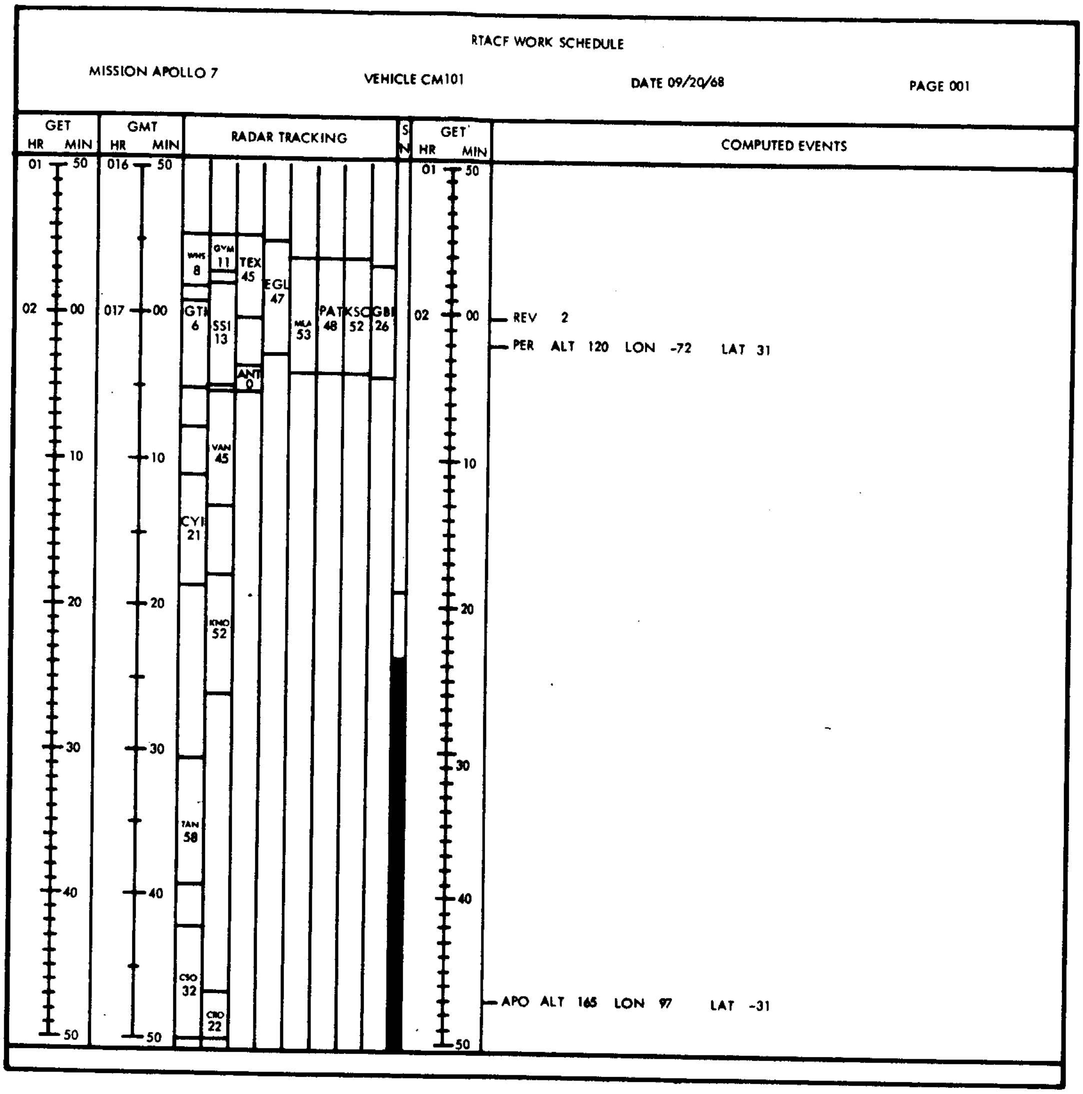

Figure 25. Work Schedule Format

TRIM AERODYNAMIC COEFFICIENTS

XCG = XXXXX YCG = XXXXX ZCG = XXXXX

WEIGHT = XXXXX

BANK ANGLE BIAS = XXXXX DEG.

MACH NO.	ALPHA	CL	CD	CL/CD
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX

Figure 26. Aerodynamics Update Summary Sheet

MODULES =	1	2 . . . .
X=		XXXXX
Y=		XXXXX
Z=		XXXXX
IX=		XXXXX
IY=		XXXXX
IZ=		XXXXX
WT=		XXXXX

Figure 27. Center of Gravity Summary Sheet

C.G., MOMENTS OF INERTIA AND TRIM ANGLES VERSUS TOTAL WEIGHT

WEIGHT LBS	X INCHES	Y INCHES	Z INCHES	IXX SLUG-FT2	IYY SLUG-FT2	IZZ SLUG-FT2	PITCH TRIM DEG.	YAW TRIM DEG.
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXX	XXX
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXX	XXX
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXX	XXX
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Figure 28. Mass Properties Summary Sheet

CSM DAP UPLINK

PRA	ADD	OCTAL	ENG. U.	UPL. U.
IXX	XXXX	XXXXX	XXXXXXX	XXXXXXX
IAVG	XXXX	XXXXX	XXXXXXX	XXXXXXX
WEIGHT	XXXX	XXXXX	XXXXXXX	XXXXXXX
PITCH TRIM	XXXX	XXXXX	XXXXXXX	XXXXXXX
YAW TRIM	XXXX	XXXXX	XXXXXXX	XXXXXXX
TLX	XXXX	XXXXX	XXXXXXX	XXXXXXX

Figure 29. Digital Autopilot Command Load Summary Sheet

SM RCS PROPELLANT COMPUTATION

	QUAD A	QUAD B	QUAD C	QUAD D
TANK	XXX	XXX	XXX	XXX
PH	XXXXX	XXXXX	XXXXX	XXXXX
T	XXXXX	XXXXX	XXXXX	XXXXX
DELT	XXXXX	XXXXX	XXXXX	XXXXX
PO	XXXXX	XXXXX	XXXXX	XXXXX
PF	XXXXX	XXXXX	XXXXX	XXXXX
MR1	XXX	XXX	XXX	XXX
MR2	XXX	XXX	XXX	XXX

	PRI	AUX	PRI	AUX	PRI	AUX	PRI	AUX
WFE	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
WOE	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
WFR	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
WOR	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
WPU	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX

Figure 30. PVT Summary Sheet

SUMMARY SHEET - PRE-BURN ENTRY

EVENT TIMES	GMT				GET			RET			LATITUDE		LONGITUDE	
	DAY	HR.	MIN.	SEC.	HR.	MIN.	SEC.	HR.	MIN.	SEC.	DEG.	MIN.	DEG.	MIN.
RETROFIRE	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
BURN TERMINATION	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
SEPARATION	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
ENTRY INTERFACE	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
BEGIN BLACKOUT (VHF)	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
BEGIN BLACKOUT (S)	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
.05G'S(EMS)	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
0.2G'S	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
REVERSE BANK	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
END BLACKOUT (S)	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
END BLACKOUT (VHF)	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
DROGUE DEPLOY	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
MAIN DEPLOY	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
LANDING	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX

BURN QUANTITIES

DELTA VELOCITY =	XXXXX	FT/SEC		
DELTA TIME =	XX	MIN	XX	SEC
WEIGHT =	XXXXX	POUNDS		
PERIGEE RESULTING =	XXXXX	N. MILES		
TRUE ANOMALY =	XXXXX	DEGREES		
ALTITUDE =	XXXXX	N. MILES		
LATITUDE =	XX	DEG	XX	MIN
LONGITUDE =	XXX	DEG	XX	MIN
EXTERNAL DELTA VX =	XXXXX			
VY =	XXXXX			
VZ =	XXXXX			

GIMBAL ANGLES	OUTER (ROLL)	INNER (PITCH)	MIDDLE (YAW)
BURN	XXX	XXX	XXX
SEPARATION	XXX	XXX	XXX
400K	XXX	XXX	XXX
300K	XXX	XXX	XXX
BACKUP BANK AT .20G	XXX	XXX	XXX
BACKUP BANK AT RB	XXX	XXX	XXX

LANDING POINTS

	LATITUDE		LONGITUDE	
	DEG.	MIN.	DEG.	MIN.
MIN LIFT	XX	XX	XXX	XX
MAX LIFT	XX	XX	XXX	XX
TARGET	XX	XX	XXX	XX
G AND N	XX	XX	XXX	XX
BACK UP	XX	XX	XXX	XX

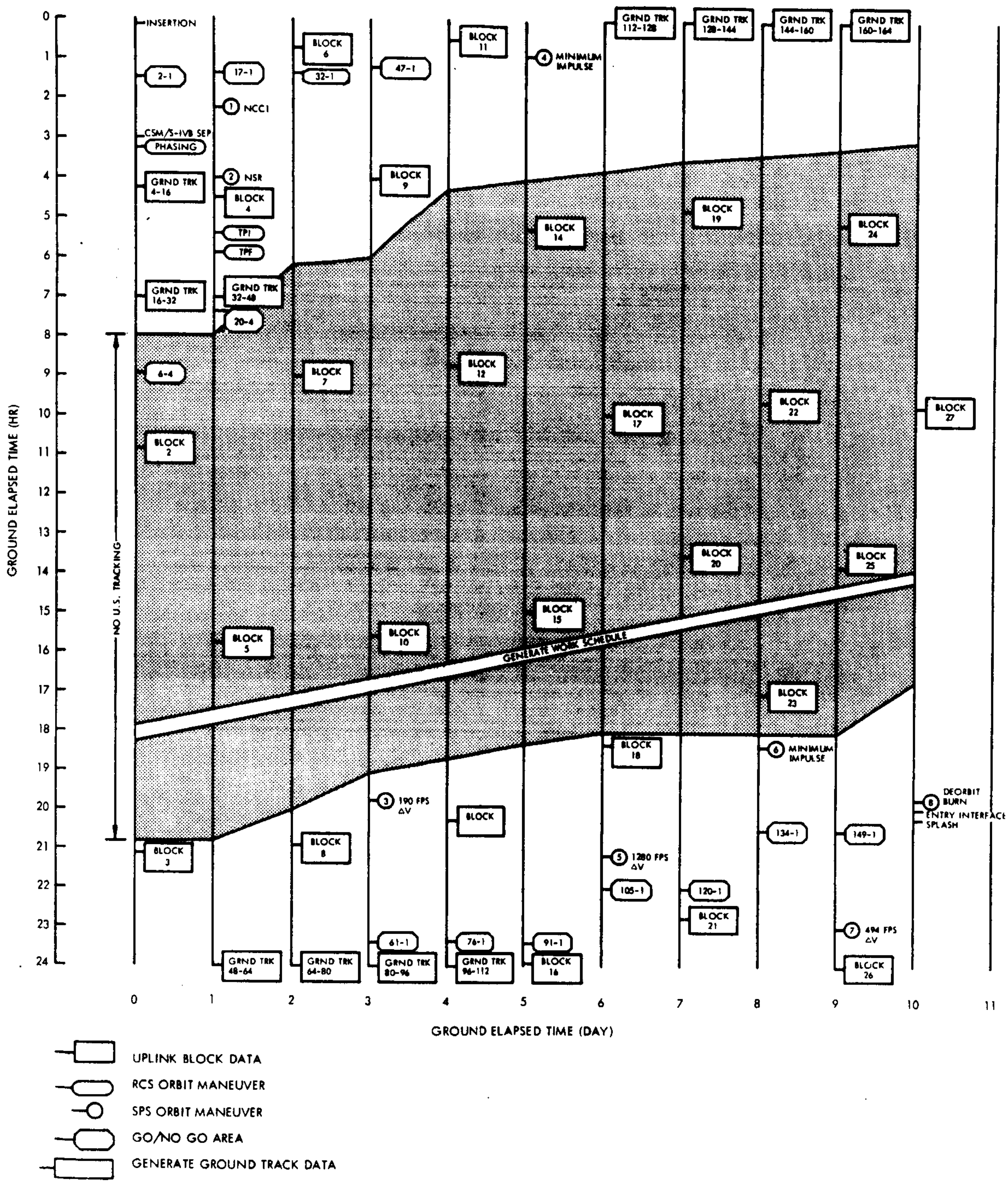
REENTRY QUANTITIES

INITIAL BANK	XXX	DEG(RIGHT)
BACKUP BANK	XXX	DEG(LEFT)
RETRB	XX MIN XX	SEC
RET400K	XX MIN XX	SEC
RET.05G	XX MIN XX	SEC
RET.2G	XX MIN XX	SEC
RET.XXG	XX MIN XX	SEC
RETBBO (VHF)	XX MIN XX	SEC
RETEBO (VHF)	XX MIN XX	SEC
RETDD	XX MIN XX	SEC
RETMD	XX MIN XX	SEC
RT400K	XXXXX	N.MILES
V400K	XXXXX	FT/SEC
GAMMA400K	XX	DEG
(RP-RT).2G	XXX	N.MILES
V.05G	XXXXX	FT/SEC
RNG.05G	XXXXX	N.MILES
RETEMS	XX MIN XX	SEC
VEMS	XXXXX	FT/SEC
RNGEMS	XX	N.MILES
MAXG	XX	
L/D	XXXX	
WEIGHT	XXXXX	POUNDS

Figure 31. ARS Summary Sheet

5. APOLLO 7 RTACF NOMINAL MISSION TIMELINE

This section of the Flight Annex presents the RTACF nominal mission timeline. The timeline is not meant to display the schedule of RTACF activities during the mission but rather to present, in graphical form, a synopsis of the events which are of particular importance in planning the RTACF work schedule, which is to be published as a separate MSC internal note. The events and their corresponding times presented in the timeline include the major nominal mission events, the region where U.S. tracking capability exists, the go/no-go decisions, block data uplink, and the generation of new ground track data. The event times used in generating the timeline were based on Apollo 7 Operational Trajectory (Revision 1).



Apollo 7 RTACF Nominal Mission Timeline

6. Apollo 7 RTACF Operational Support Team

The following individuals will man positions in the Flight Dynamics Staff Support Room and the Auxiliary Computing Room during the Apollo 7 mission.

Flight Dynamics Staff Support Room

	1st Team	2nd Team	3rd Team
Trajectory Support Chief	C. E. Allday (FAB)	S. D. Holzaepfel (FAB)	L. D. Davis (FAB)
Assistant Trajectory Support Chief	E. R. Hischke (FAB)		H. Garcia Jr. (FAB)
Prelaunch IP Specialist	S. R. Newman (FAB)		
Reentry Specialist	D. W. Heath (LLB) O. Hill (LLB)		
Rendezvous Specialist	K. A. Young (OMAB)		
Maneuver Specialist	R. R. Regelbrugge (OMAB)		
Engineering Aide	P. A. Moore (ITT) G. G. Arrington (ITT)	E. D. Langford (ITT)	P. L. Lankford (ITT) S. L. Hallmark (ITT)

Auxiliary Computing Room

	1st Team	2nd Team	3rd Team
ACR Chief	C. D. Chenoweth (TRW)	R. D. Davis (FAB)	P. A. DiValerio (TRW)
ACR Engineers	D. C. McDougall (TRW) B. G. Schneider Jr. (FAB) T. L. Turner (FAB)	V. R. Dragotta (TRW) H. L. Sanders (TRW) J. H. Kawasaki (TRW)	L. Baker Jr. (TRW) D. W. Sager (TRW) W. R. Pruett (FAB)
Program Consultant	J. F. Bem (LEC)	K. Grimsley (LEC)	W. H. Smiers (LEC)
Run Coordinators	D. R. Wiggins (LEC) J. A. Kleinjan (LEC)	D. Womack (LEC)	J. E. Friebele (LEC) B. N. Ferguson (LEC) G. E. Mohring (LEC)
Engineering Aide	C. M. Toole (ITT)		C. S. Trigg (ITT)

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